CR-160295

SPACE CONSTRUCTION SYSTEM ANALYSIS

TASK 2 FINAL REPORT

SYSTEM ANALYSIS OF SPACE CONSTRUCTION

JUNE 1979

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LINE ITEM 3

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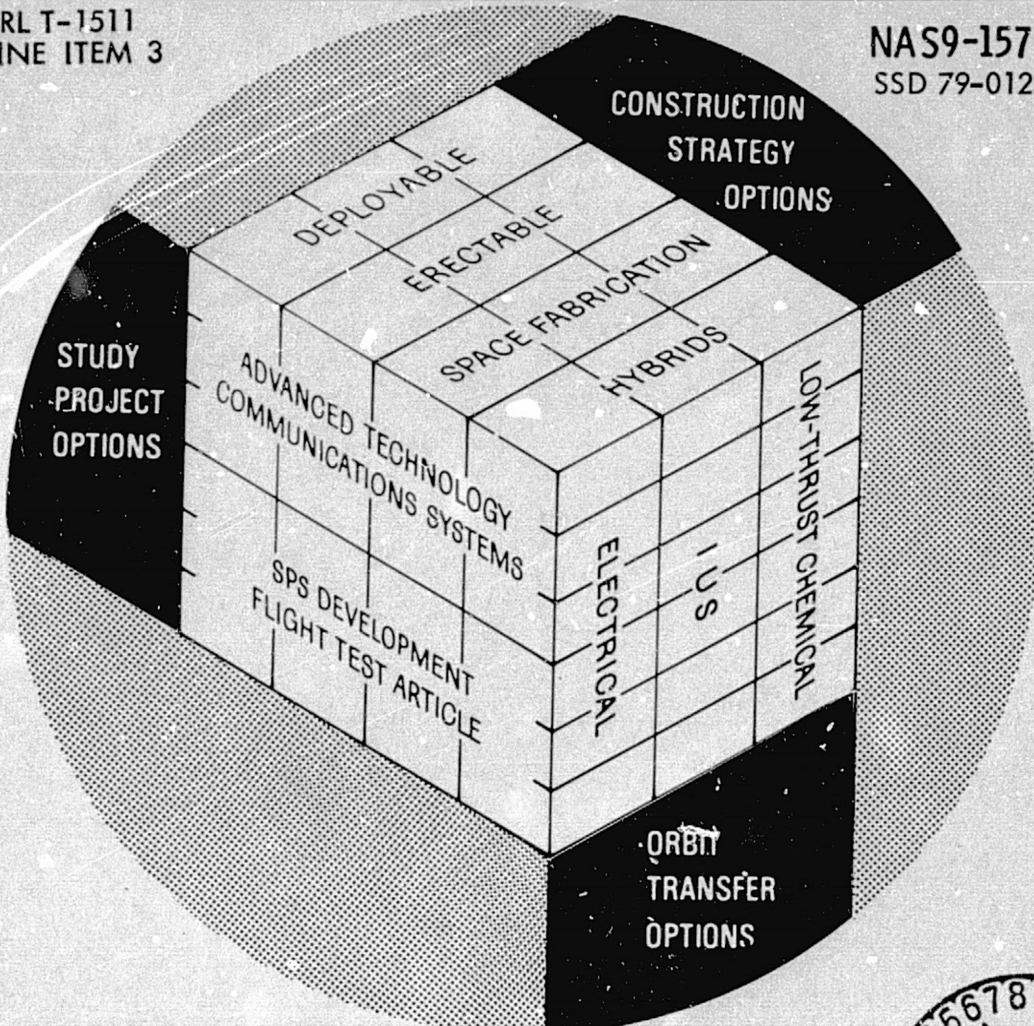

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SPACE CONSTRUCTION SYSTEM ANALYSIS STUDY

TASK 2 FINAL REPORT -

SYSTEM ANALYSIS OF SPACE CONSTRUCTION

Contract No. NAS9-15718
DRL T-1511
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June 1979

Approved by
Ellis Katz

Rockwell International
Space Division

FOREWORD

This report summarizes the results of several systems analysis studies of space construction projects, primarily dealing with areas of space construction support services, construction facilities, orbit altitude and orbit transfer. The document is a study product of Task 2, System Analysis of Space Construction, Contract NAS9-15718, Space Construction System Analysis Study. This contract effort was conducted by the Satellite Systems Division, Space Systems Group of Rockwell International Corporation, for the National Aeronautics and Space Administration (NASA), Johnson Space Center (JSC).

The study was performed under the direction of Ellis Katz, Study Manager. The following persons made significant contributions toward completion of the analyses reported herein.

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Major documents resulting from Part I of the contract effort are listed below:

Space Construction System Analysis,
Project Systems and Mission Descriptions,
Task 1 Final Report, SSD 79-0077
April 26, 1979

Space Construction System Analysis, Task 2 Final Report -
System Analysis of Space Construction, SSD 79-0123,
June 1979

Space Construction System Analysis, Task 3 Final Report -
Construction System Shuttle Integration, SSD 79-0124,
June 1979

Space Construction Data Base
SSD 79-0125, June 1979

Space Construction System Analysis, Special Emphasis
Studies Final Report, SSD 79-0126, June 1979

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1.0 INTRODUCTION

This report presents the system analyses of space construction which were performed as Task 2.0 of the contract study. Task 1.0 (SD 79-0077) defined three project systems to be used as models for the investigation of alternative construction methods and processes. Task 2.0, then, contains the construction method derivations and supporting analyses required for construction system definition. Specifically, Task 2.0 applies the construction requirements and strategies for each project system to (1) define construction scenarios, (2) identify critical construction functions, and (3) synthesize alternative methods for each function. The resulting methods definitions were organized and formatted into a construction data base where key ideas and knowledge of resource requirements and constraints for individual construction situations can be easily extracted and applied to new project systems. The data base is published as a separate loose-leaf volume (SSD 79-125) for user convenience and to easily accommodate future additions from new construction studies.

Complementary to the central task of defining alternative construction methods, important interrelated issues necessary to construction system definition were investigated. These include construction support services, construction facility implications and the impacts of various orbit transfer propulsion modes on space construction. This report contains the results of these complementary investigations.

Section 2.0 summarizes the project system design definitions used in the construction analyses and outlines the methods definition process. The organization and content of the Construction Data Base is also briefly discussed.

Section 3.0 presents analyses of individual support services issues. These include: attitude stabilization and control during construction and between construction flights, thermal control of sensitive elements during construction, illumination and TV services including visual (bright source) interference effects and power/illumination relationships and preliminary implications of individual electrical power demands and their impacts on space construction processes.

Section 4.0 presents construction facility considerations highlighting the potential benefits of using the Space Operations Center as a construction base. Facility arrangements/characteristics which reduce or eliminate some of the limitations and constraints associated with building out of the orbiter are shown along with their improved productivity potential.

Section 5.0 presents construction orbit trades and orbit transfer analyses. Factors affecting the selection of construction orbit altitude are discussed and the minimum safe altitude for the construction of each project system is defined. Propulsion thrust, performance and sizing characteristics covering chemical, solid and solar electric propulsion concepts are presented for LEO to GEO orbit transfer missions. Thrust loads, T/W impacts and TVC/structural stiffness interactions with lightweight space structures are discussed. Trades of alternative techniques for delivering propulsion modules to the construction orbit are also presented.

2.0 CONSTRUCTION ANALYSIS

This section summarizes the construction analysis process beginning with the project system definitions and construction requirements from Task 1 of the study and leading to the compilation of alternative construction methods in a construction data base (SSD 79-0125). A major factor permeating this analysis process was the need to confine and focus the efforts on only the most important issues. The number of possible combinations of all problem variables (project configurations, construction strategies, construction functions, construction equipment/aids, and construction procedures) cannot be treated in a single study. To reduce the number to a manageable level, the number of project systems was held to three, and only the most critical and most representative construction functions were selected for detailed methods analysis. The three project systems selected for the study are briefly summarized in Section 2.1. The stream-lined construction analysis process is outlined in Section 2.2, and the construction data base resulting from these analyses is described in Section 2.3.

2.1 PROJECT SYSTEM DESCRIPTIONS

This section contains a brief description of the SPS test article and the two advanced communications platforms on which construction analyses were performed. The Task 1 final report (SSD 79-0077) contains a more detailed description of these projects.

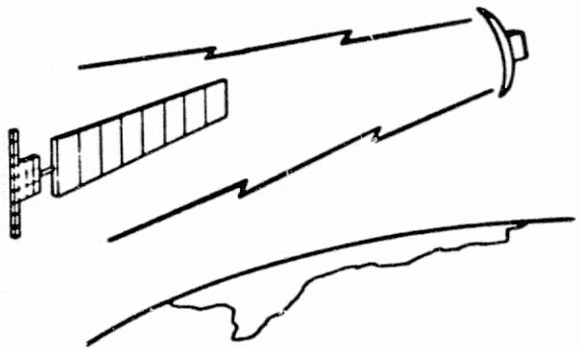
2.1.1 SPS Test Article Description

The scenario for the SPS test article specifies a 1985 time period as currently planned in the SPS Red Book. The prime objective of this project is to perform space-to-space microwave tests.

Operating, servicing, and growth features of the overall project scenario are noted on Figure 2.1-1. Within the operating scenario, a relatively high orbit, at least 550 km (300 nmi) altitude, is envisioned due to the very low ballistic coefficient (W/C_DA) $\approx$ 1.0) of this configuration. The flight vehicle must also be capable of initially adjusting its orbit and stationkeeping with a co-orbiting rectenna.

The servicing scenario shows manned servicing for LEO operations and unmanned remote servicing concepts above LEO.

The general arrangement of the SPS test article is represented in Figure 2.1-2. This configuration represents the GEO operational configuration which includes the installation of the Solar Electric Propulsion (SEP) orbit transfer engine modules. This configuration, therefore, contains all of the systems that must be addressed in the construction analysis task.



PROJECT SCOPE

1985 TIME PERIOD
μ-WAVE TESTS (PRIME)
VERSATILE, MULTI-USE FACILITY

OPERATING SCENARIO

ORBIT: ≈300 NMI, i = 28.5°
FLT MODES: ORBIT ADJUST/STATIONKEEP
μ-WAVE φ CONTROL
μ-WAVE THERMAL EFFECTS
DAY - NIGHT OPS
MISSION DURATION: SPS LEO TEST ≈6 MO

SERVICING SCENARIO

MANNED SERVICING AVAIL, LEO
UNMANNED-REMOTE SERVICING ABOVE LEO
RESUPPLY STATIONKEEPING PROPELLANT
REPLACE FAILED/DEGRADED MODULES

GROWTH SCENARIO

ADV TECHNOLOGY ORBIT, ≈ 400 NMI
REOUTFITTED: VARIETY OF POSSIBLE USES,
INCREASED ENERGY STORAGE
ADDED USER INTERFACE PROVISIONS
ORBIT TRANSFER, UP TO GEO

CONFIGURATION/DESIGN IMPACTS

- SINGLE DOF SOLAR ARRAY
- "RELATIVE" STATIONKEEPING
- ALL MAIN MODULES ACCESSIBLE, REVERSIBLE INSTALLATION
- "TEST BED" INSTRUMENTATION
- ENERGY STORAGE SIZED TO HOUSEKEEPING
- PROVISIONS FOR ADDED ENERGY STORAGE

Figure 2.1-1. SPS Test Article Scenario

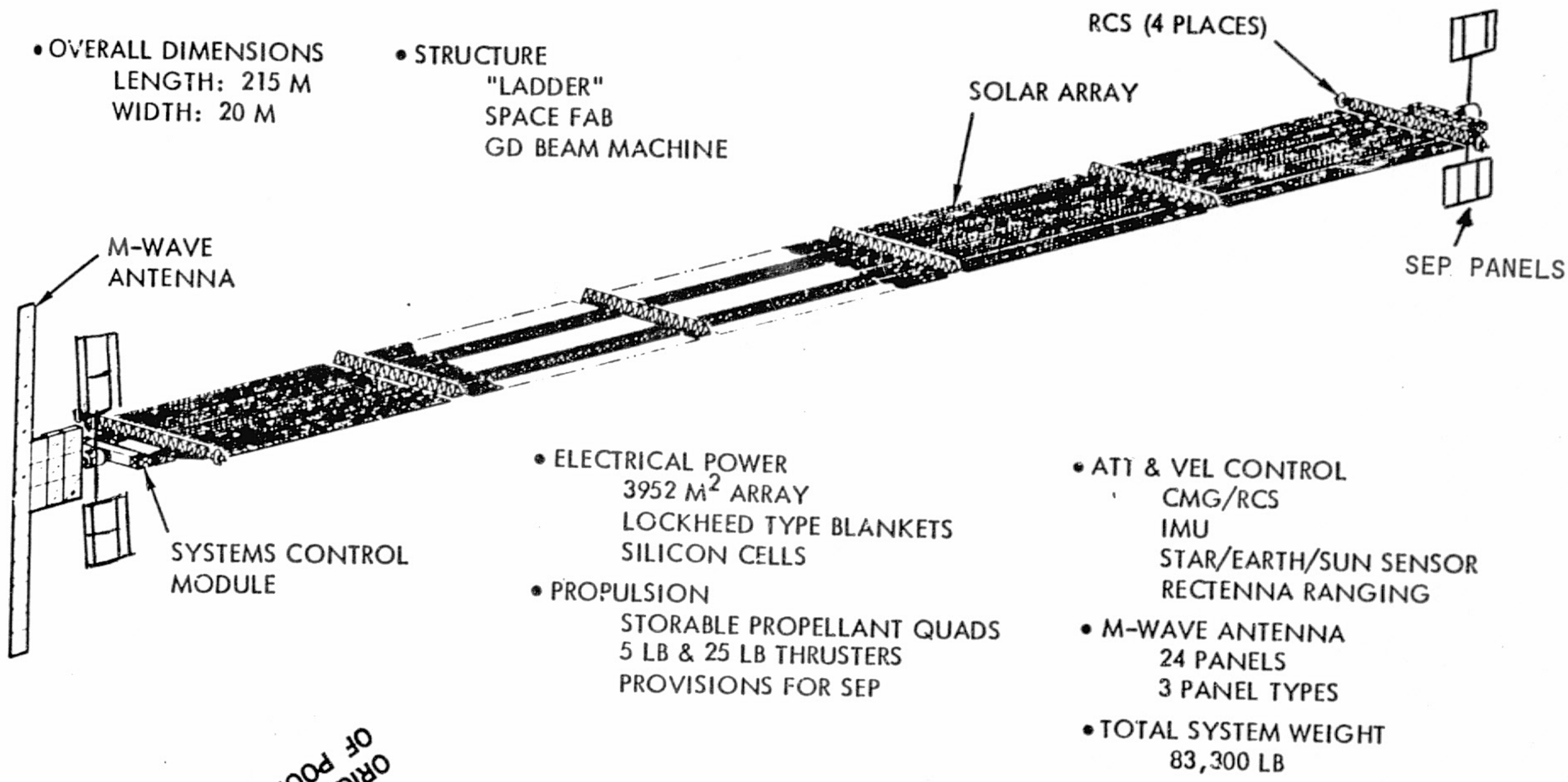


Figure 2.1-2. SPS Test Article Configuration

The SPS test article project consists of a ladder-type structural arrangement utilizing space-fabricated beam members to which 25 solar blankets are attached. The ladder structure is an assembly of beams fabricated by a single beam builder in orbit. The beam configuration is that developed by General Dynamics SCAFE study with modifications as required, such as increased cap gauges and diagonal cord diameters.

All of the larger modular items such as the RCS modules and the systems module are attached to the structure via berthing ports. The berthing port concept is the three-petal, neuter concept, baselined for the Shuttle orbiter. Smaller units such as the electrical junction boxes and the solar blanket switching boxes will be secured to the structure with clamp-type devices that are compatible with the structural beam configuration and load capability. The clamping devices that secure the solar array switching boxes also provide the attachments for the individual solar array blankets. Electrical lines are secured to the structure with special clips. The clips require pre-punched holes in the post members of the fabricated beams.

The systems module which contains the electrical power storage batteries and controls, the CMG's, the TT&C equipment, and the heat-rejection radiator is also the structural bridge that provides the structural interface between the solar array structure and the rotary joint to which the microwave antenna is attached. A similar structural bridge at the opposite end of the solar array structure provides the support for the rotary joint and the solar electric propulsion modules used for growth mission orbit transfer.

The solar array consists of 25 solar blankets. Each blanket is attached to the transverse beams of the structure. The attachment is provided with clamp-type fittings to which the solar blankets are attached at three places along the 4-m-width of the blanket. Power leads plug into individual switching boxes. From each of the switch boxes power lines run along the longitudinal beams to interface with the systems housing and continue on to the power slip ring of the rotary joint.

The rotary joint provides one degree-of-freedom rotation between the solar array and the microwave antenna. It also provides the support for the SEP modules. The rotary joint as a unit is attached to the systems housing via a berthing port. A berthing port also is provided on the other end of the rotary joint unit to accept the microwave test antenna or other test articles.

When growth mission operations to GEO are desired, as discussed in the project scenario, the SEP modules will be installed. Each of four modules contains a mounting post which is designed to plug into the rotary joints making the structural attachment as well as the electrical power and data control connections. Two additional modules are mounted to two of the module/post configurations to make two 24-engine clusters which are required at the microwave antenna end of the SPS microwave test article. The rotary joint required at the other end of the solar array structure will be installed when the orbit transfer mode is desired.

The estimated weight of the SPS microwave test article in the LEO operational configuration is 37,800 kg (83,160 lb). The orbit transfer configuration estimated weight is 49,200 kg (108,250 lb).

2.1.2 Advanced Communications Platform Descriptions

The advanced communications platform scenario, summarized in Figure 2.1-3, calls for its introduction in the 1990 time period. The concept outlined here would employ several frequency bands, each utilizing high-density frequency reuse techniques to greatly enlarge the communications capability associated with a single orbiting platform.

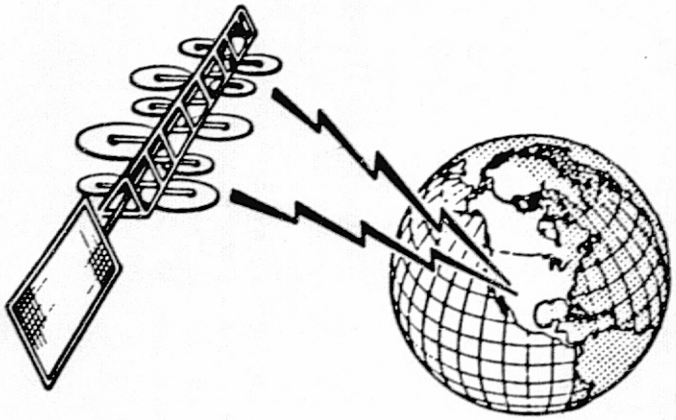
The operational system will be placed in GEO with good access to the U.S. The platform concept is applicable to and could grow to be global in nature with additional platforms placed at other locations satisfying traffic needs in other areas. The operating system requires relatively precise pointing and stability to maintain the desired multiple beam pattern coverage of the U.S. Both N-S and E-W stationkeeping are required to hold the narrow assigned slot in the congested GEO orbit and to eliminate major antenna pointing excursions which would be required without stationkeeping. The large investment represented by this high-capacity platform concept would likely call for at least a 20-year service life. Also, to maintain the very high level of communications services (99.98% dependability) we've grown to expect, it is envisioned that service should not be interrupted for stationkeeping maneuvers, sun occultation periods, and (possibly) routine servicing operations.

Two structural arrangement concepts were developed to implement the communications platform requirements discussed above. Figures 2.1-4 and 2.1-5 illustrate these two concepts. Except for minor variations, such as the arrangement of the antennas and the attaching concept of the system control module, most of the subsystems are identical between the two concepts. The variations are a result of the different structural arrangement concepts.

2.1.2.1 Configuration Description—Erectable Communications Platform

The antenna platform concept illustrated in Figure 2.1-4 consists of an erectable-type structure assembled of tapered struts with ball end fittings engaging receptacle-type unions. The platform is boosted to geosynchronous orbit with low-thrust chemical-fueled engines. The 16 antennas are arranged in two groups (1) eight 4-6 GHz C-band receivers and transmitters, and (2) eight 12-14 GHz K-band receivers and transmitters. Growth capability for additional antennas is also provided.

During orbit transfer the solar arrays are folded parallel to the longitudinal axis of the platform which is also the direction of acceleration. Each antenna horn and boom support is also retracted during the orbit transfer mode. The reflector portion of each antenna, however, is in the deployed position.



PROJECT SCOPE

1990 TIME PERIOD
SATISFY PROJECTED GROWTH FOR CURRENT
SERVICES & INTRO NEW SERVICES
TELEPHONE/TELECONFERENCE, VIDEO....

REDUCE GEO CONGESTION
COULD BE GLOBAL SYSTEM

CONFIGURATION/DESIGN IMPACTS

- EPS SIZED FOR CONTINUOUS GEO OPS INCLUDING OCCULTATION PERIODS
- ACCURATE BEAM POINTING/STABIL INCLUDING DURING STATIONKEEP
- ACCESS FOR SERVICING & GROWTH
- LOCATIONS FOR ADDED ANTENNAS
- PROVISION FOR ADDED ELEC POWER

OPERATING SCENARIO

GEO ORBIT—USA ACCESS
STABILIZED EARTH POINTING
EW & NS STATIONKEEPING
UP TO 20 YRS SERVICE LIFE
PROVIDE UNINTERRUPTED SERVICE

SERVICING SCENARIO

UNMANNED/REMOTE
5-7 YR SERVICE INTERVAL
RESUPPLY STATIONKEEPING PROP
REPLACE FAILED/DEGRADED MODULES

GROWTH SCENARIO

EXPAND CAPACITY, POSS NEW SERVICES
ADD 18-30 GHZ ANTENNAS, NEW MODULATION ELEC
ADD SPACE-TO-SPACE LINK(S)
ADD ELEC POWER

Figure 2.1-3. Advanced Communications Platform Scenario

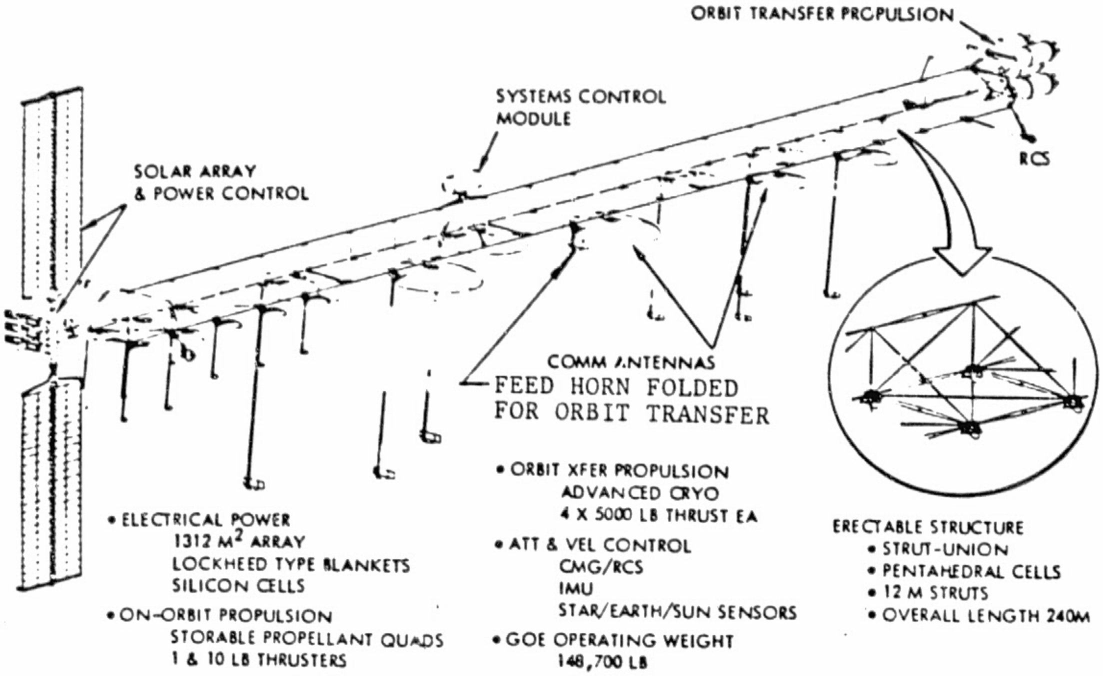


Figure 2.1-4. Erectable Communications Platform Concept

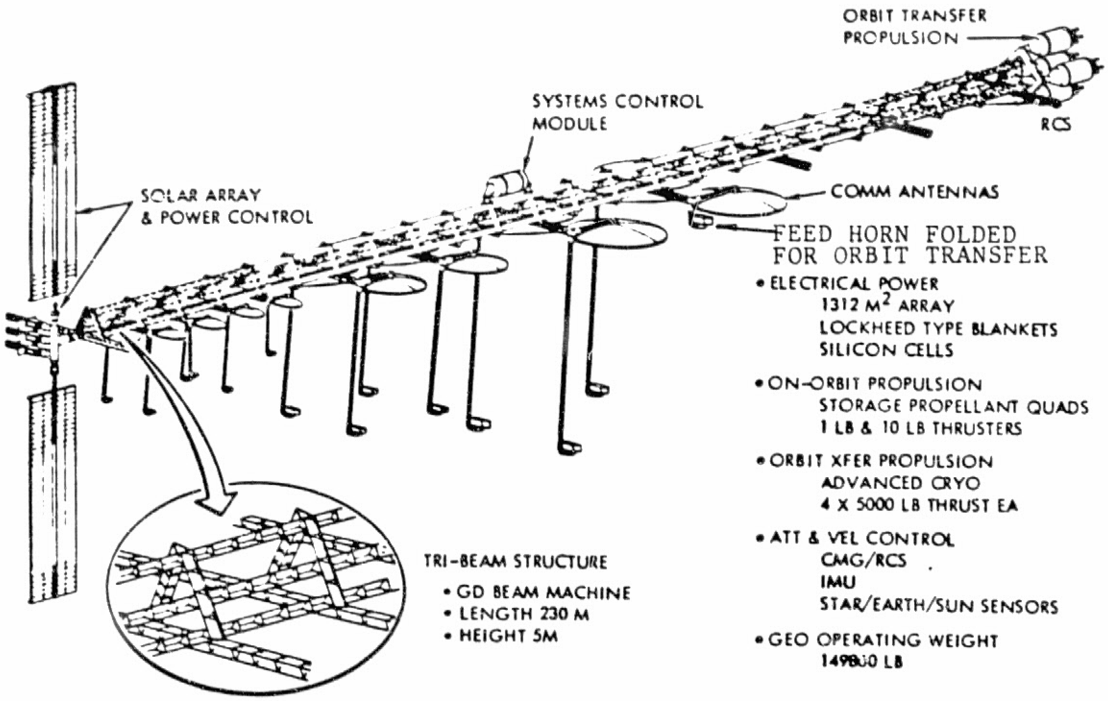


Figure 2.1-5. Space-Fabricated Communications Platform Concept

The platform structure consists of double tapered tubes with ball-type end fittings. The tubes are formed from two conical tubes joined at their large ends. Most of the tube assemblies are joined to each other through a receptacle type of union member, creating a pinned joint. However, the antenna mounting concept utilizes strut ends and receptacles that are designed to transmit moments. The support arrangement for the RCS pods, the systems module, and the orbit transfer propulsion modules utilize struts arranged to form A-frame reaction members. Most of the struts are a common length and size. However, the two load conditions described above use unique struts to fulfill their individual requirements. The struts are assembled into a linear, pentahedral structural arrangement.

All of the larger modular items such as the antennas, the GN&C/ATT&C module, the orbit transfer propulsion, and support structure are attached to the structure via berthing ports. The berthing port concept is the three-petal, neuter concept baselined for the Shuttle orbiter.

Smaller units such as the electrical junction boxes are secured to the antenna-mounting unions. The electrical lines are secured to the struts with clamping-type wire-supporting clips.

The solar arrays are mounted to a rotary joint which provides a 360° rotation capability perpendicular to the orbit plane. A 24° nodding capability is also provided to permit full sun illumination during all sun declination angles. A folding capability for orbit transfer is also provided.

The battery power storage system, which is sized to provide continuous operation during the orbit eclipse periods, is packaged into three independent units. Each package of batteries includes the battery chargers and controls, thermal control insulation and meteoroid protection, and its own heat-rejection radiator system. Each unit is a replaceable item.

The rotary joint provides for the power transfer from the power generation system to the platform through a slip ring assembly.

A system module containing the GN&C CMG's and sensor, the TT&C receivers, transmitters, antennas, etc., and a central data/signal processor is provided in a centrally located position on the platform. Thermal control, meteoroid protection, and heat-rejection radiator systems are provided as part of the module to support these systems.

A communications message switching control unit is centrally located within the C-band antenna complex and a similar unit is also centrally located within the K-band antenna complex.

The last items to be installed will be the orbit transfer propulsion modules. The propulsion modules attach to the supporting structure utilizing berthing ports to effect the joint and to establish the lines interfaces.

The complete platform less the propulsion modules has an estimated weight of 60,500 kg (133,400 lb).

2.1.2.2 Configuration Description—Tri-Beam Space-Fabricated Platform

Many features of the tri-beam space-fabricated platform (Figure 2.1-5) are similar if not identical to those of the erectable concept. Consequently, this description will concentrate on those features that are unique to this concept.

This concept represents an antenna platform utilizing a space-fabricated structure with a low-thrust chemical-fueled orbit transfer system. The 16 antennas are arranged in two groups: (1) eight 4-6 GHz C-band receivers and transmitters, and (2) eight 12-14 GHz K-band receivers and transmitters. Growth capability for additional antennas is also provided.

During orbit transfer, the solar arrays are folded parallel to the longitudinal axis of the platform which is also the direction of acceleration. The antenna horn and boom support is retracted during the orbit transfer mode. The reflector portion of the antenna remains in the deployed position.

The platform structure consists of members fabricated in orbit by a single beam builder and assembled by use of appropriate fixtures. The individual beam configuration and the beam builder device are from the General Dynamics SCAFE study concepts.

The installation of the larger modular units utilizes the berthing port concept. The description of this installation concept is identical to that discussed for the erectable antenna platform concept.

Smaller units such as the electrical junction boxes will be secured to the structure with clamp-type devices that are compatible with the structural beam configuration and load capability. The electrical lines are secured to the structure with special clips. The clips require pre-punched holes in the post members of the beams.

The electrical power generation system, including the solar arrays and the power storage battery arrangement, and the rotary joint through which the electrical power is transmitted to the antennas and subsystems, are identical to the concept description for the erectable platform.

The systems module contents and installation concept are identical to that of the erectable platform, as are the communications message switching control units.

The last items to be installed will be the orbit transfer support structure and the orbit transfer propulsion modules. The support structure interfaces with the three longitudinal members of the platform structure by means of berthing ports. The propulsion modules attach to the supporting structure in the same manner.

The complete platform, less the propulsion modules, has an estimated weight of 61,000 kg (134,200 lb).

2.2 CONSTRUCTION ANALYSIS PROCESS

Construction analysis is an iterative process, as pictured schematically in Figure 2.2-1, and will require more than a single cycle through the functions indicated to arrive at acceptable design solutions. Rockwell experience in this field suggests that, to develop the appropriate level of detail in construction analysis, it is necessary to specify the geometry, dimensions, materials, interfaces, and other physical properties which define the construction requirements. Thus, it is necessary to generate a concept for the construction fixture which would define the situation in terms of the location of the work and the access to the work. Indeed, the concept of the fixture, as has been earlier suggested, needs to be integrated with the definition of the project system to assure that what is designed can be practically constructed. With the system definition and the construction fixture concept in mind, it is then possible to devise a construction strategy which would define in what sequence or order the various construction tasks would be undertaken. As an example, it would be necessary to understand whether the structure is fully completed before equipment is installed, or whether equipment is installed as the structure is built. The answer to these issues can impact considerably the construction methods that might be candidates for the task.

The core of the construction analysis process is to consider each construction task/function and devise alternative methods for executing that function. These alternatives may differ in degree of automation, in the detailed sequence of operations, in utilization of various construction equipment or aids—or in impact upon the project system design or construction fixture. After having examined the alternative methods which could be used to perform each of the construction functions, it is then possible to examine the entire array of construction functions in an end-to-end context and to select those methods for each function which would, in the aggregate, produce the most effective and economic employment of the materials, construction support equipment, and the resources of the orbiter/crew to perform the operations. The final step in this cycle is to define the design of the construction fixture, the construction support equipment, and the orbiter/construction facility provisions. These definitions could, in turn, reflect back upon the project system design, upon the construction strategy, and even upon the construction methods which could lead to further iterations of the cycle.

A basic four-step process was applied to produce the Construction Data Base which was the Part I study objective. The methods evaluation/selection and integrated construction system design steps in the above iterative process will be performed in Part II of the study. The 4-step process applied here is:

- Step 1—Define a construction fixture concept for each project system. The concept definitions are based on trade evaluations of alternative fixture arrangements with respect to the orbiter along with the configuration features of the projects to be constructed.

- Step 2—Determine a construction strategy (general sequence of construction operations) for each project system. The purpose of this step is to establish explicit construction scenarios for the identification of "critical functions."

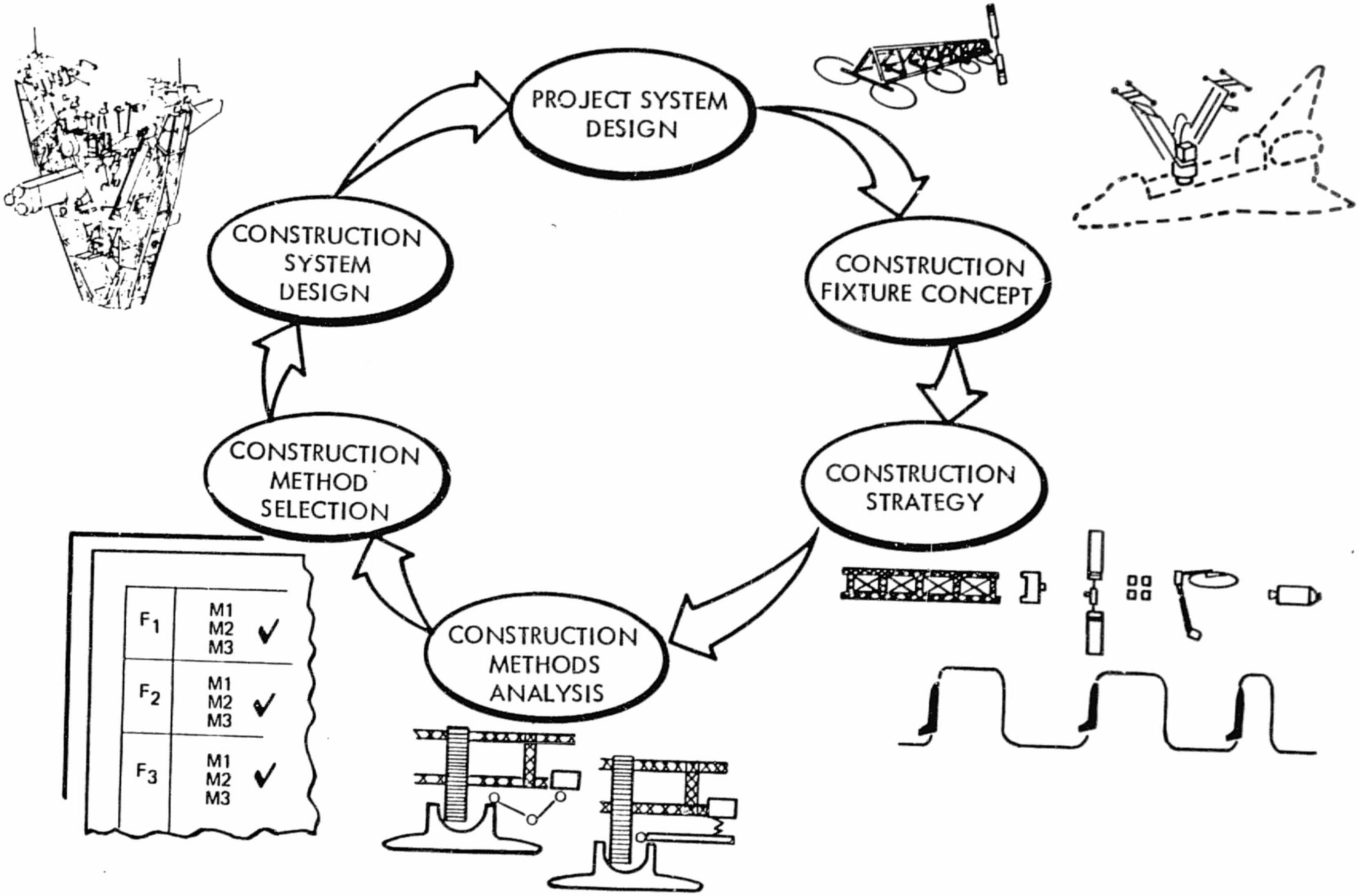


Figure 2.2-1. Iterative Construction Process

- Step 3—Using the fixture concepts and strategies from the preceding steps, each project construction was "walked through" to determine those construction functions which were considered to be "critical." Critical functions are those which can significantly impact the construction support equipment requirements, the operations, and/or the project system design.

- Step 4—Analyze each critical function to drive out requirements and alternative construction methods. The objective is to define alternative methods which are matched to specific construction scenarios, are practical, and which are representative of fundamentally different approaches.

Each of these steps is further explained in the following paragraphs.

2.2.1 Construction Fixture Concepts

The project construction fixture concept plays a significant role in the construction of any large space project. As such, in most applications, it is designed specifically for a particular project. The three study projects were sufficiently different in their structural and systems installation approach to require unique construction fixtures.

Each fixture must have the capability to support the project during all phases of construction and assembly. It must also provide project translation capability and orbiter revisit capability. EVA activities must be accommodated with adequate restraints, etc. Finally, the fixture must be packageable within the orbiter payload bay.

In addition to these requirements, three design issues are identified:

- The orientation of the construction fixture with the orbiter
- Performing the construction activity within the local vicinity of the fixture
- Space fabrication construction with a single or multiple beam builder

These requirements and issues were addressed in a trade study that defined the construction fixture concepts for each of the study projects. This trade study is presented in Appendix A of this report. A brief description of each of the construction fixtures follows.

2.2.1.1 SPS Test Article Fixture

The construction fixture concept for the fabrication and assembly of the SPS test article is illustrated in Figure 2.2-2. The fixture consists of structure to which the test article retention arms, beam positioner, and rotational handling device are mounted. The rotational handling device supports

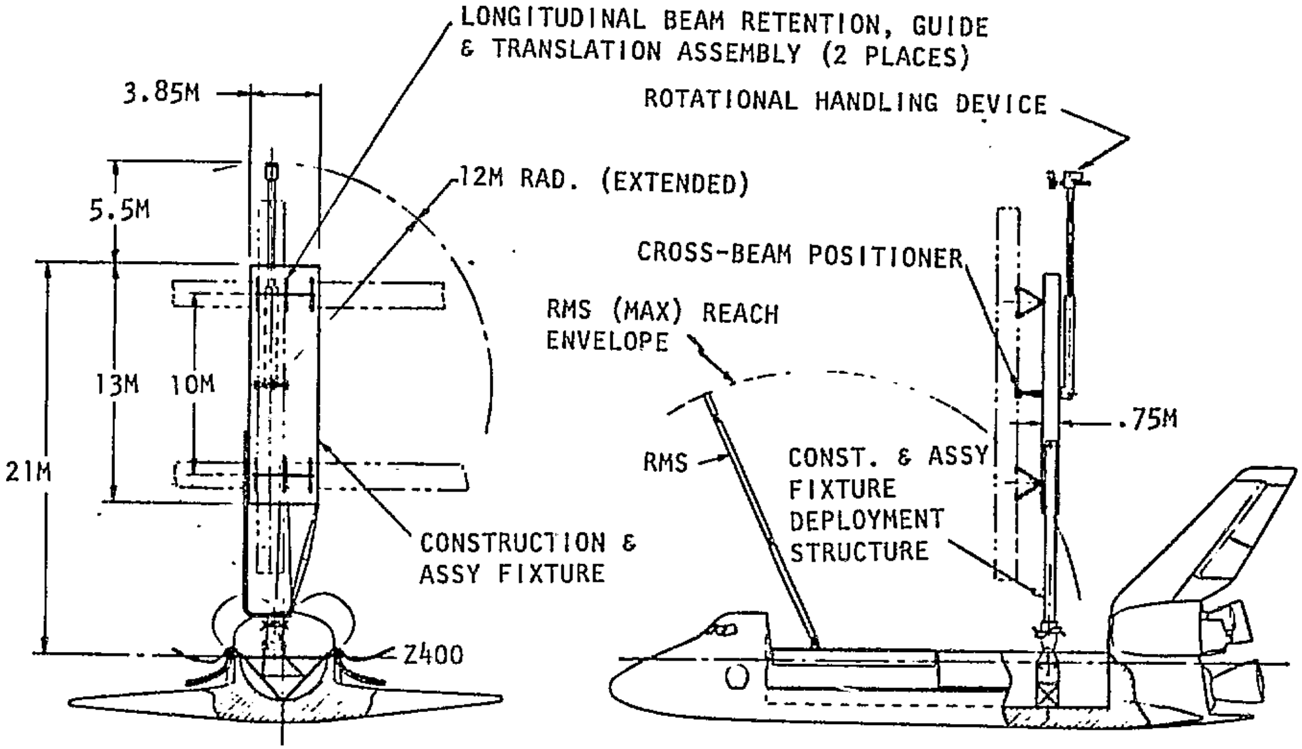


Figure 2.2-2. SPS Test Article Construction Fixture Concept

the beam builder during fabrication, and also supports other special construction devices. The test article translation is accomplished by providing articulation of the retention arms which permits the cross-beams to be "stepped" through the retention arms during the translation operations.

The total construction fixture is attached to the orbiter via a berthing port and appropriate structural members to raise the fixture to permit translation of the completed SPS test article.

2.2.1.2 Erectable Advanced Communications Platform Construction Fixture

The construction fixture developed for the assembly of the erectable communications platform concept is shown in Figure 2.2-3. The fixture consists of a single post/guide rail that supports the translation cradle. The guide rail/translation cradle assembly is supported from the orbiter. The translation cradle supports struts in their proper relationship during assembly and also provides the capability to translate the total platform the distance of one pentahedral bay. Platform supporting clamps secure the platform to the upper end of the support post, permitting the translation cradle to release the platform and return to the assembly location

The thrust structure support cradle locates and supports the thrust module attach tripods in their proper relationship. A rotation capability of thrust structure support cradle permits the assembly of the thrust module support pods to within the reach envelope of the orbiter RMS.

2.2.1.3 Space-Fabricated Advanced Communications Platform Construction Fixture

The construction fixture for the tri-beam structure is illustrated in Figure 2.2-4, and provides the support and location of the beam builder during fabrication, the support and translation capability of the platform, the location of the cross-beams, and the provisions for the attachment of the cross-beams to the longitudinal beams via welding.

The translation of the project system is accomplished by providing articulation of the holding arms, thus permitting the cross-beams to be stepped through the holding arms during the translation operation.

Cross-beam positioning devices accept the fabricated beams from the RMS and precisely locate the beams for attachment. After the tri-beam structure has been completed the beam builder support arm and the beam positioner support structure are removed, thus clearing the fixture for the installation of the subsystems.

2.2.2 Construction Strategy Development

Initial construction strategies were developed for each of the study projects described in Section 2.1. However, prior to the determination of these specific strategies, two fundamental strategy approaches or construction principles) were briefly examined. The results of this investigation are somewhat interrelated to the fixture concept definitions above and, thus, are presented first. The individual strategies for each of the project systems then follow.

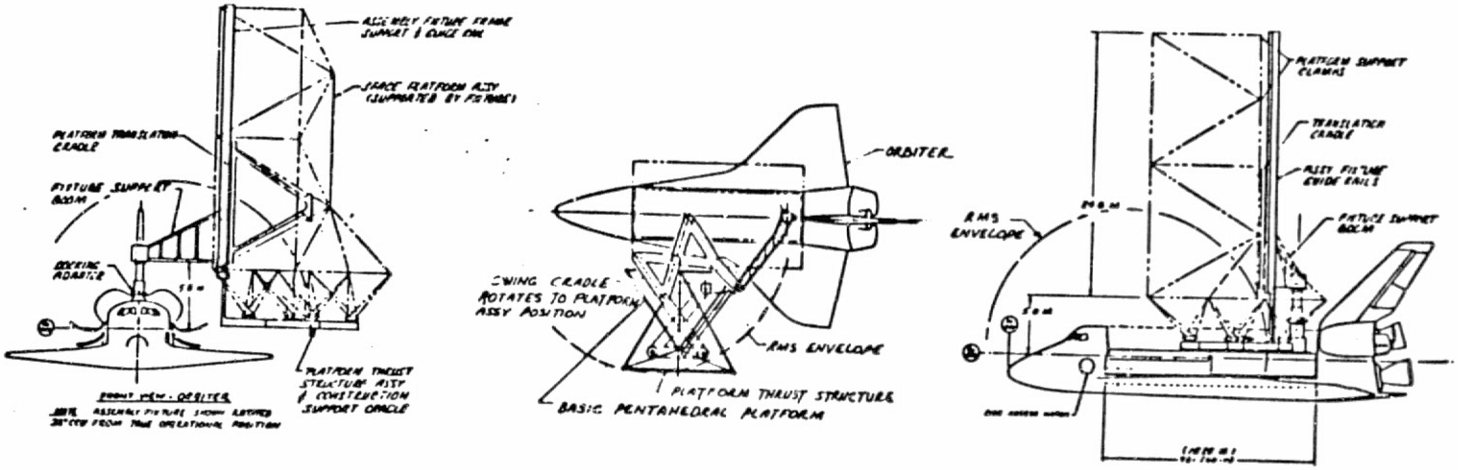


Figure 2.2-3. Erectable Communications Platform Construction Fixture Concept

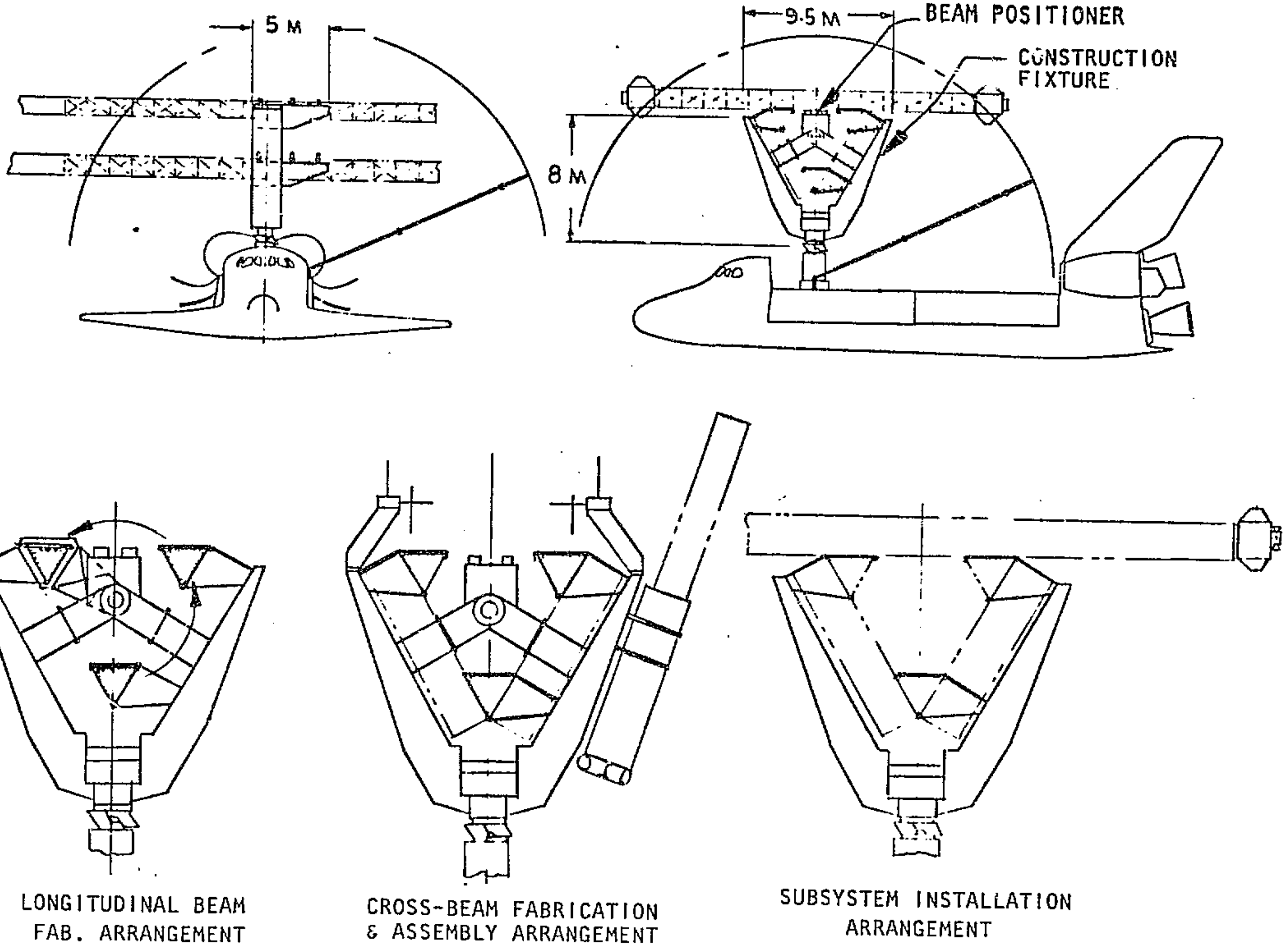


Figure 2.2-4. Space-Fabrication Communications Platform Construction Fixture Concept

Two fundamental strategy approaches were identified: (1) serial construction, where various types of construction processes are done serially; and (2) combined construction, where several types of construction processes are done simultaneously.

The first option follows the more conventional method of construction, i.e., foundation or structure; then, utilities and subsystems—in that order. The second option constructs and assembles the structure, installs the utilities and subsystems complete for a particular area of the project, and then moves to the next area, completing it, etc., until the total project is completed. Figure 2.2-5 schematically illustrates these two options as applied to the space-fabricated advanced communications platform.

Option 1 requires the capability to translate the project through the construction fixture in order to assemble each subsystem on the total project. Consequently, an added fixture complexity is imposed. The translation capability, however, provides a desirable degree of flexibility in construction not only for the planned construction sequences, but also to better accommodate any unplanned anomalies or contingencies. The flexibility to reconfigure the construction fixture to the most efficient arrangement to accommodate a particular installation sequence is also provided with this option. Figure 2.2-6 illustrates this capability as applied to the space-fabricated antenna platform. The capability to complete the installation of a particular subsystem that may require special equipment and operations and then reconfiguring for the following installation task appears to have the potential for more efficient construction operation and better productivity.

The potential also exists with Option 1 of providing a more efficient cargo packaging capability by having more like materials packaged and by not being as restrictive in the cargo removal sequence, as may be the situation when utilizing the second option. Figure 2.2-7 illustrates this potential.

The construction fixture of Option 2 would appear to be less complex than that required for Option 1, but would require all the fixtures and special equipment required for subsystems handling and installation to be available at the same place at the same time. This requirement could also create a complexity equal to or even exceeding the translation requirement of Option 1. Figure 2.2-8 illustrates a construction fixture arrangement that might be required to implement the construction of the space-fabricated antenna platform by the Option 2 method.

The strategy approach for Option 1 was selected for this study. Consequently, the primary structure is fabricated and assembled, first followed by the installation of the utilities and subsystems, with the more sensitive subsystems being the solar array blankets for the SPS test article and the antennas of the communications platforms. The discussion of this approach and its implications on construction for each of the three projects follows.

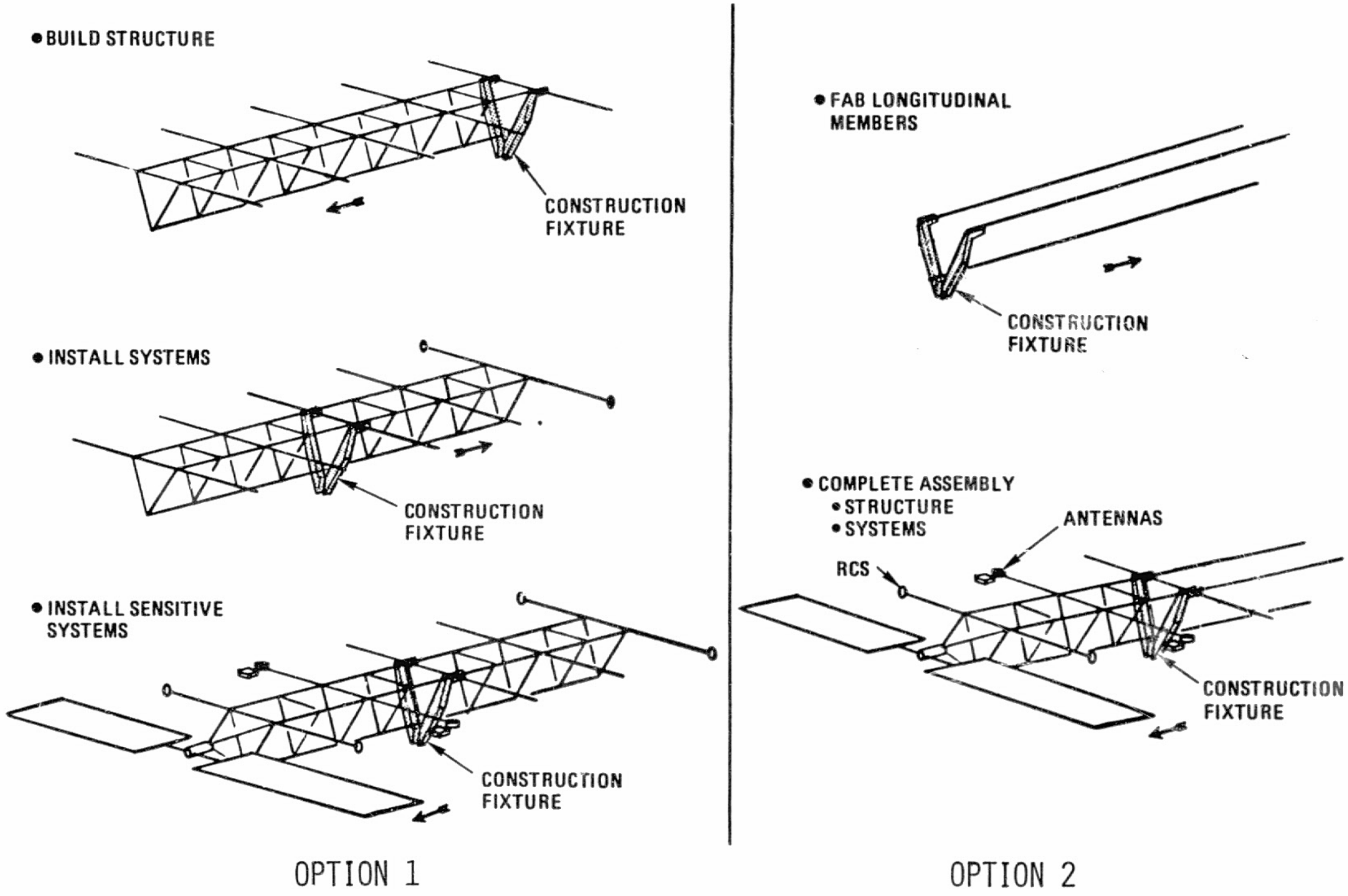
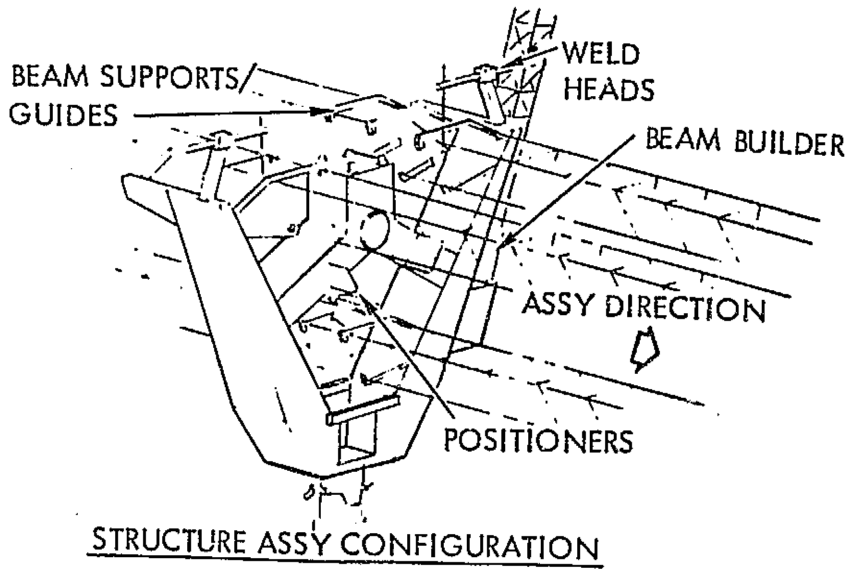


Figure 2.2-5. Construction Strategy Options



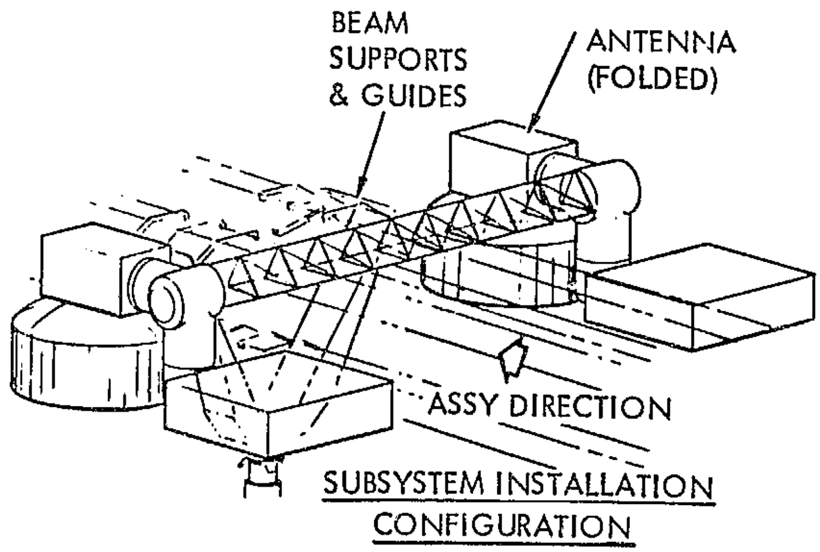
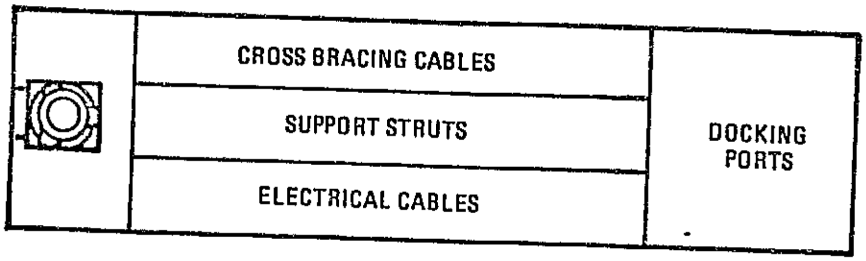
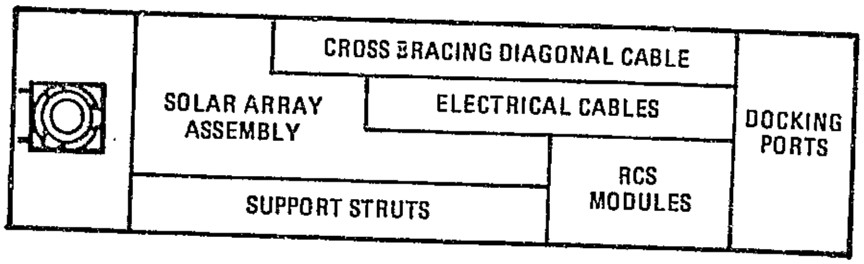


Figure 2.2-6. Construction Fixture Reconfiguration (Option 1)



OPTION 1



OPTION 2

Figure 2.2-7. Payload Bay Packaging

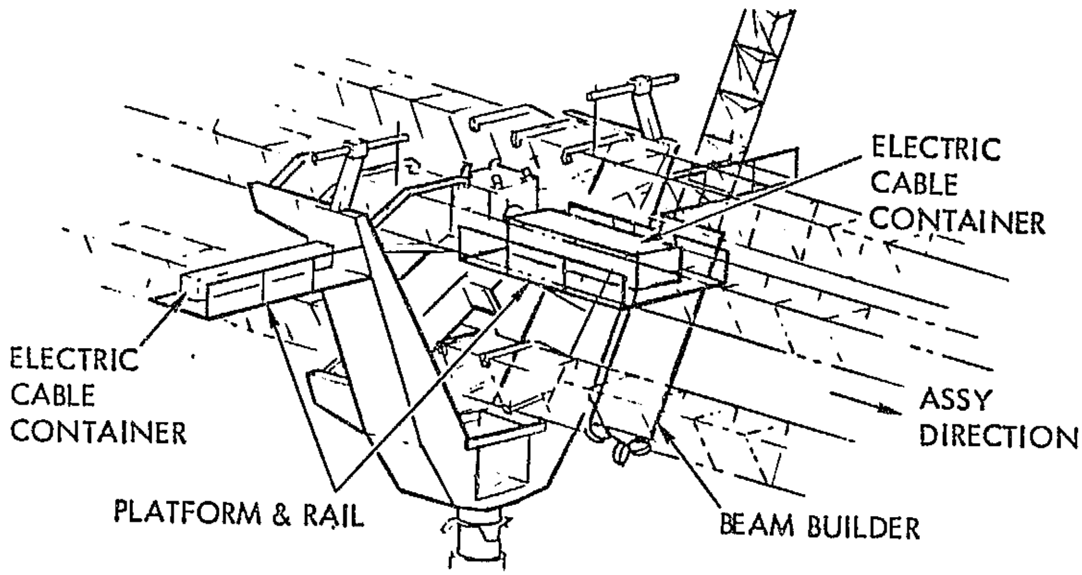


Figure 2.2-8. Construction Fixture Arrangement (Option 2)

SPS Test Article Construction Strategy

The construction strategy is schematically illustrated in Figure 2.2-9 for the assembly of the SPS test article. As previously stated, the construction fixture has the capability to translate the project so that the construction activities always occur in the vicinity of the fixture. Four translations of the project are indicated to complete the operational configuration shown. The implications of the construction support services such as power, illumination, etc., and the implications of construction attitude control and the berthing/docking operations are discussed in Section 3.0

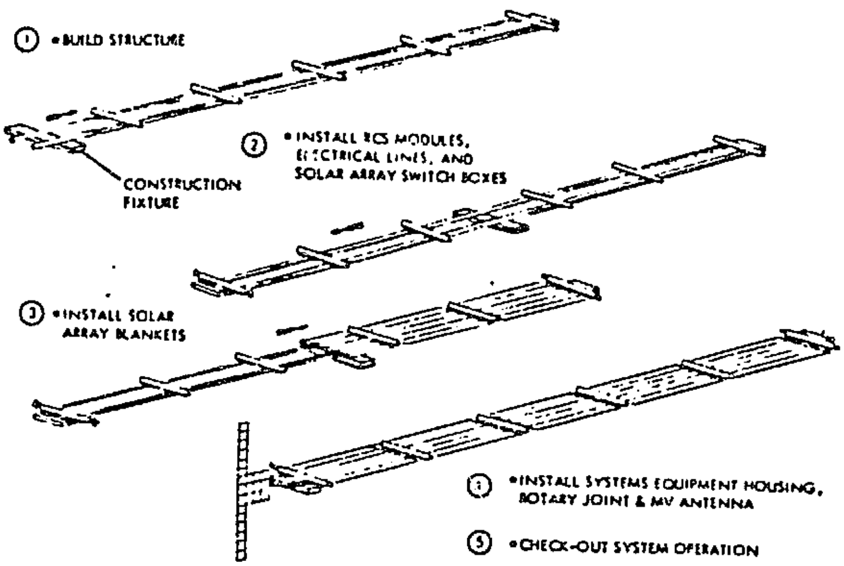


Figure 2.2-9. Construction Sequence—SPS Test Article

The construction sequence shown was developed so that the construction fixture would be located adjacent to the microwave antenna at the completion of the assembly. This arrangement permits the fixture to be used as a berthing port for the installation of the microwave antenna. The fixture can also be stored in this position for future use in servicing or changeout of the payload (microwave antenna).

Erectable Advanced Communications Platform Construction Strategy

The construction strategy for this platform concept is illustrated in Figure 2.2-10. The construction sequence follows the concept of assembling the structure, the installation of utilities and subsystems, and completing the assembly with the installation of the antennas. The project is completed with the fixture being adjacent to the orbit transfer engine mounting structure. This position permits the installation of the propulsion modules on the project from the orbiter. Three complete translations are indicated for the complete assembly of this erectable antenna platform concept.

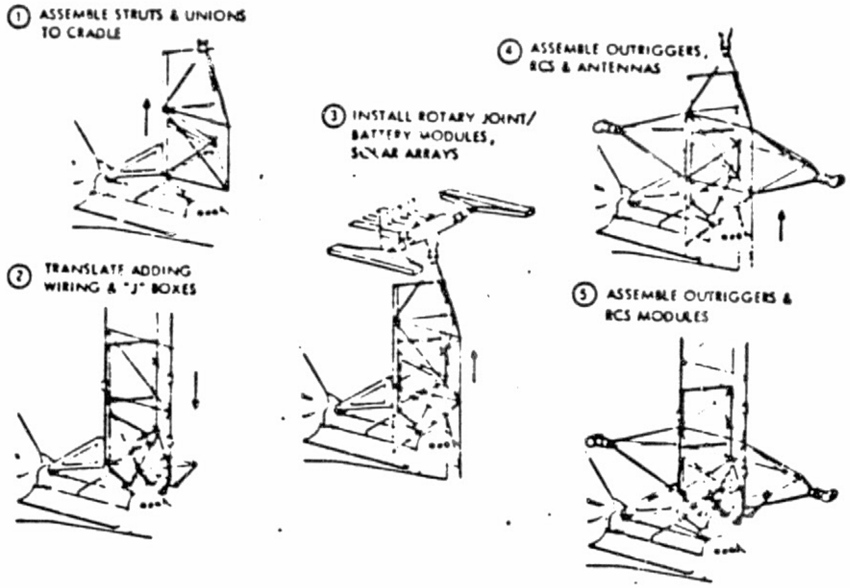


Figure 2.2-10. Construction Sequence—Erectable Advanced Communications Platform

Space-Fabricated Advanced Communications Platform Construction Strategy

The construction strategy of this antenna platform concept is schematically illustrated in Figure 2.2-11. The construction sequence developed for this

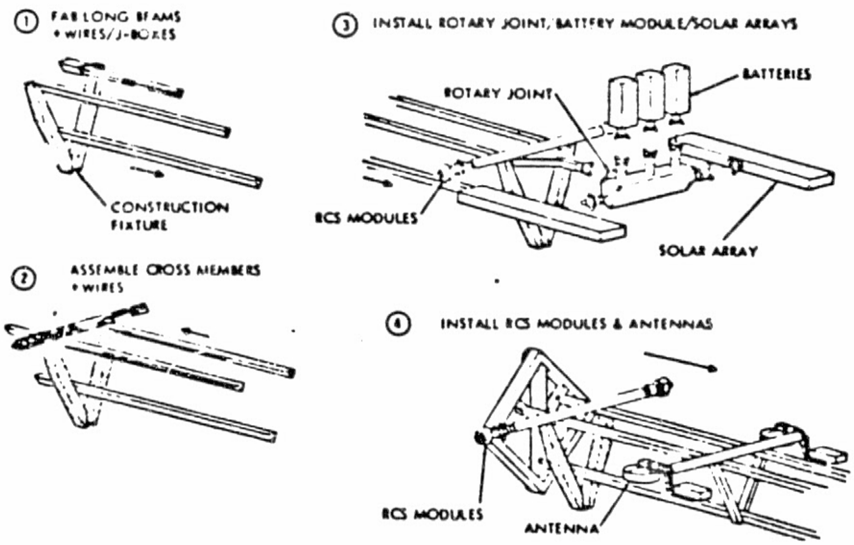


Figure 2.2-11. Construction Sequence—Space-Fabricated Advanced Communications Platform

project varies slightly from the basic philosophy of fabricating and assembling the structure first. In this sequence the electrical lines are installed on the longitudinal beams as they are being fabricated. The electrical lines are also installed on the cross-beams before the beams are joined to the longitudinal members. This construction sequence deviation allows one less translation to occur. The sequence, therefore, only requires two translations for completion of this platform concept.

The construction fixture in this concept is also located adjacent to the orbit transfer engine mounting structure at the completion of the project assembly. This arrangement permits the use of the fixture as a Shuttle orbiter berthing port for the installation of the propulsion modules.

2.2.3 Critical Functions Identification

Having defined construction fixture concepts and appropriate construction strategies for each of the three project systems, construction functions which must be performed in the construction of each project can be identified. Generic construction functions are shown in Figure 2.2-12 with correlation checkmarks indicating what functions typically apply to the various types of project system elements. The application of thse generic functions to all elements of the project system, in the proper order and in the appropriate circumstances, results in a completed system ready for orbit transfer to its mission orbit.

Thirty-six such function were originally identified for the three project systems. This number was reduced to 22 "critical" functions by eliminating those which were basically redundant between projects, thus resulting in a more manageable number of functions for subsequent methods identification and development.

By "critical functions" we mean those functions which may produce significant impacts on any or all of the following factors: project system design, construction equipment/operations, and technology requirements. These impacts could range from the sizing of structural elements and/or overall configuration dimensions to the requirement for special construction equipment and aids, either of which could lead to major technology requirements in support of their development programs.

The critical functions identified for the SPS test article are shown in Figure 2.2-13. The following discussion highlights the nature of each construction problem and its importance to the overall process in determining the most important critical functions for the SPS project. The 11 critical functions identified for this case are numbered (Figure 2.2-13). In the case of the docking ports, (1), the problem is to gain access to the ends of the transverse beams and install the ports into the fragile space-fabricated beams. In the case of the RCS modules, (2)/(3), the problem is to make the installations upon the structural assembly with sufficient reach/access for making the mechanical and electrical connections. The installation of attachment fittings, (4), to the structure (for subsequent attachment of power switch boxes and solar blankets) requires detailed operations to install these relatively small devices upon the beams. At each end of the solar array blankets there will

ITEM \ FUNCTION	DEPLOY	FABRICATE	TRANSPORT	POSITION	JOIN	INSTALL	CONNECT	SERVICE	QA
STRUCTURAL ELEMENTS	√	√	√	√	√	√	X	X	√
ASSEMBLIES	√	X	√	√	√	X	X	X	√
WIRING/ LINES	√	X	√	√	X	√	√	X	√
MODULES	√	X	√	√	X	√	√	√	√
BLANKETS/ MEMBRANES	√	X	√	√	X	√	X	√	√
SYSTEM	√	X	X	X	X	X	X	√	√
COMPONENTS	X	X	√	√	X	√	√	X	√

Figure 2.2-12. Generic Construction Functions

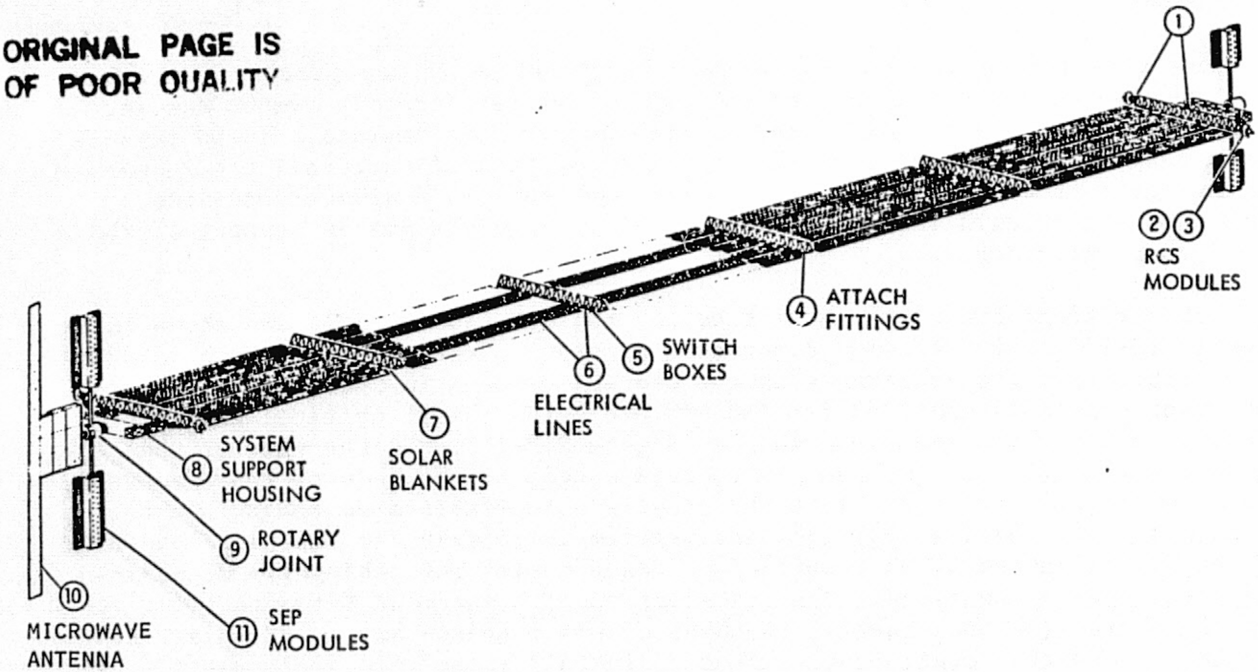


Figure 2.2-13. Critical Functions—SPS Test Article

be switch boxes, (5), to control the power from each blanket assembly. The installation of these switch boxes requires special attacnments/connections with the structure and the solar array blankets. The installation of electrical lines, (6), represents heavy cable bundles which could be difficult to handle/deploy from the cargo bay and attach to the space-fabricated structure. The solar blankets, (7), represent a case of tensioning a surface between structural members, and will require special methods to deploy and tension the blankets. The unique issue with the system support housing, (8), is to install a large module which requires two-point installations. The rotary joint installation, (9), represents the special problem of a double-ended docking device which must be installed within relatively tight tolerances. The microwave antenna, (10), requires the translation of a very large module from the cargo bay to the SPS assembly, and a multiplicity of power and data connections. The solar electric propulsion modules, (11), are planned for subsequent installation in a later phase of the test program and represent a potential post-construction servicing operation.

The above type of thinking was also applied to the advanced communications platform projects. The resulting critical functions for these projects are presented in Figures 2.2-14 and 2.2-15 for the erectable and space-fabricated configurations, respectively.

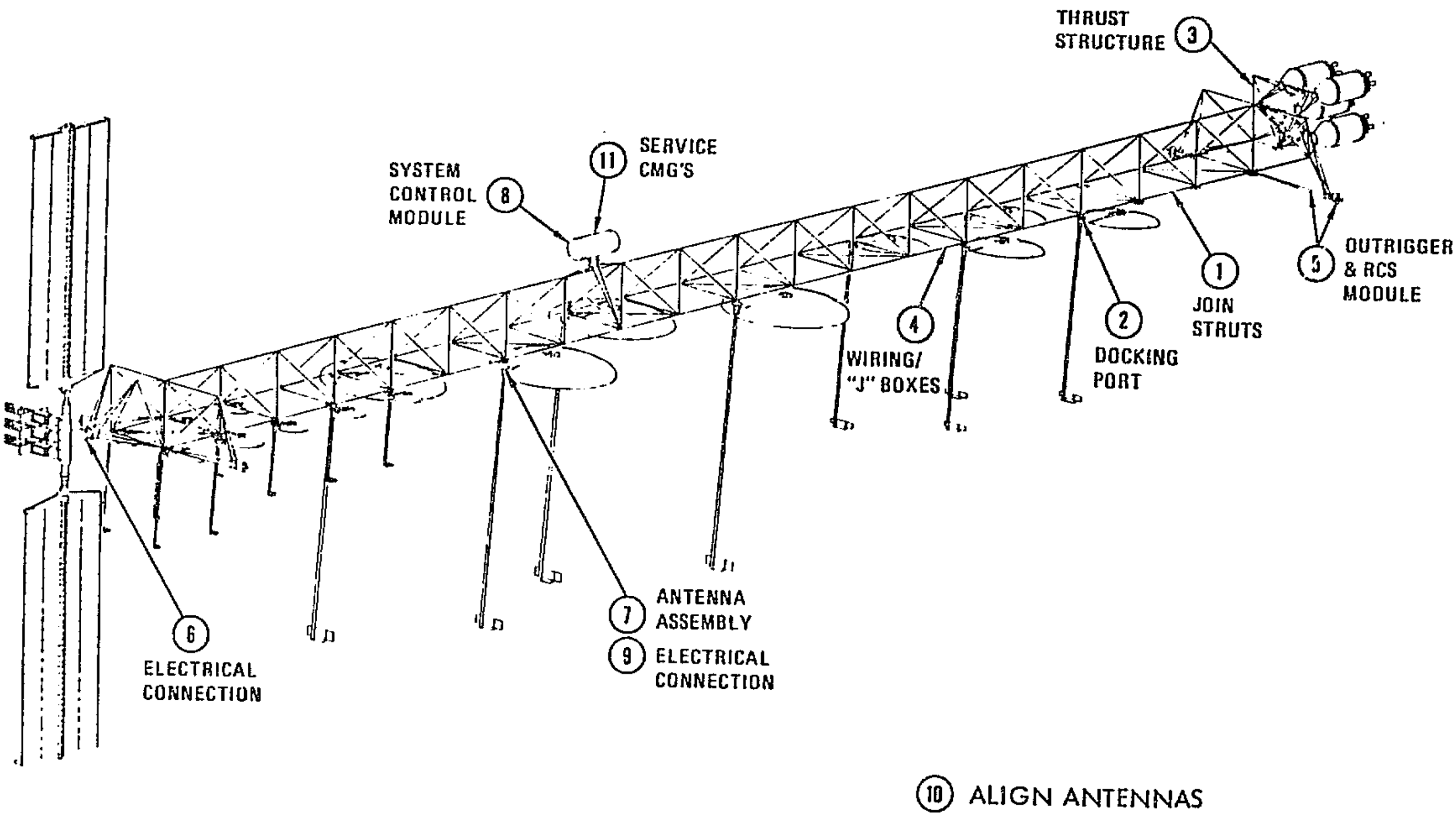


Figure 2.2-14. Critical Functions—Erectable Communications Platform

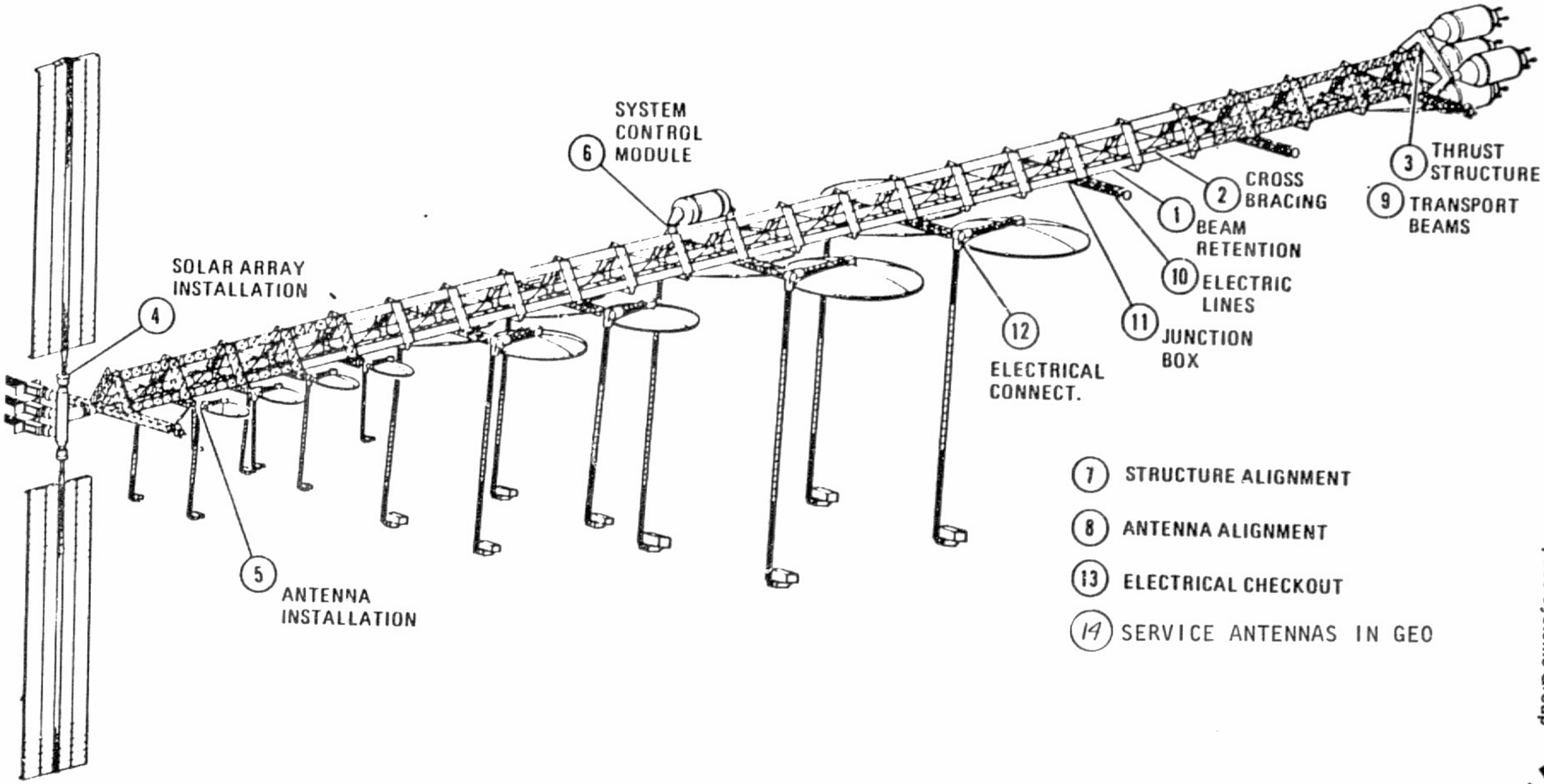


Figure 2.2-15. Critical Functions—Space Fabricated Communications Platform

2.2.4 Construction Methods Analysis

The fourth and final step in the construction analysis process for this part of the study is the identification and definition of alternative construction methods. The potentially large numbers of possible methods which could be conceived created the need for a process which focuses only on the most important and most viable alternatives. To meet this need, a preliminary methods definition and screening activity was introduced into the overall methods analysis process as shown in Figure 2.2-16.

Each of the critical functions from the previous step was assigned to a project engineer (PE) who then had the responsibility for carrying the methods analysis through to its end point. The end product of the process was two fundamentally different ways to accomplish the construction operations implied by the critical function. Both manual and automatic (mechanized) modes were considered with several ways identified for each. A simple preliminary description for each method concept was prepared, considering the physical situation which involved the partial state of construction of the platform and the location and nature of the parts to be added.

These preliminary concepts were presented before an internal engineering review board (ERB) by the PE. The ERB effort resulted in the selection of two or three ways to be more fully defined for inclusion in the data base. In a number of the ERB's, new methods were formulated and/or modifications were introduced to the preliminary concepts. The selection of those methods identified for further definition was generally based upon the desire to provide a good cross-section of viable construction techniques which would be useful in future space construction analyses (including Part II of this study).

Upon conclusion of the ERB, the PE then initiated a more detailed definition of each selected method. Each method and its related circumstances established requirements for the various construction equipment types to be used in the execution of that method. These data and the construction equipment characteristics were used to develop operational sequences and timeline segments. These, in turn, permitted determination of the required construction support services and the generation of resource profiles for the alternative methods, thus providing basic comparative data.

Emphasis during this analysis activity was on the following questions:

- Does the identified method represent a fundamental solution?
- Does it represent a general class of space construction problems?
- Does it impose special or unique requirements on either the construction equipment or the project system design?
- Can the construction equipment used in this method be used in other methods?
- How does the particular assumed construction situation affect the candidate methods?
- What are the special circumstances that lead to the importance of a given method?

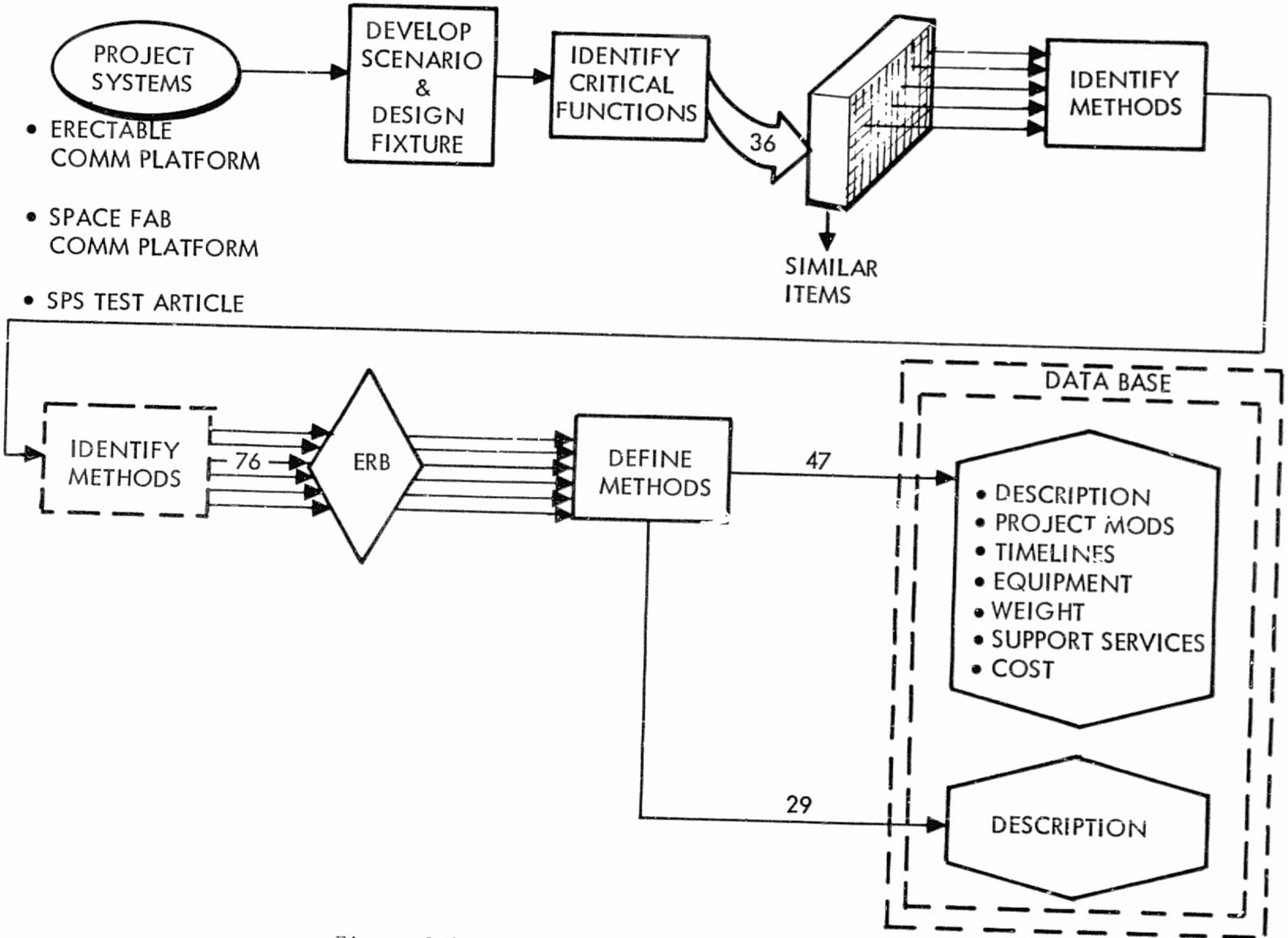


Figure 2.2-16. Data Base Preparation Process

This type of thinking was applied throughout the methods analysis tasks to always maintain a focus on the most important construction issues. The resulting individual methods data were then formatted into coded pages, as depicted in Figure 2.2-17, for inclusion in the Construction Data Base.

A total of 76 methods was identified for 22 critical functions. The selection process of choosing two or three of the alternate methods identified in the first step of the process resulted in 47 methods being defined in the data base. Descriptions for each of the remaining 29 methods were also included.

2.3 CONSTRUCTION DATA BASE

The preceding construction analysis process produced 47 construction methods definitions. These were compiled and formatted into a Construction Data Base (SSD 79-0125), which is the main product of Part I of the study.

The data base is organized to permit the addition of data from future studies. The contents are coded to permit unlimited additions and convenient access to the information by generic project type (space-fabricated, erectable or deployable). The data base is divided into four major sections:

- Project Systems Description (Section I)
- Construction Methods (Section II)
- Construction Support Equipment (Section III)
- Indexes (Section IV)

2.3.1 Project Systems Description (Section I)

This section contains a brief description of each of the three project systems which were the basis for the information contained within. Sketches of the important subsystems/major components, and construction scenarios (strategies) are also included so that the user can understand the context in which various construction methods are applied.

2.3.2 Construction Methods (Section II)

This section is the core of the data base as it contains the basic information concerning construction methods and is indexed by the generic construction process, function, and item as described in Figure 2.3-1. Since the understanding of what constitutes an "Assembly" and other items can vary, Table 2.3-1 lists the definitions as used in the data base for each of the "Items."

A review of the design, construction scenario, and initial construction fixture concept for each of the three projects resulted in the identification of 22 critical functions or operations (e.g., How do we install the system control module?). While these operations were identified considering a specific design and construction strategy, they are expected to be representative of the major operations to be performed in any construction process.

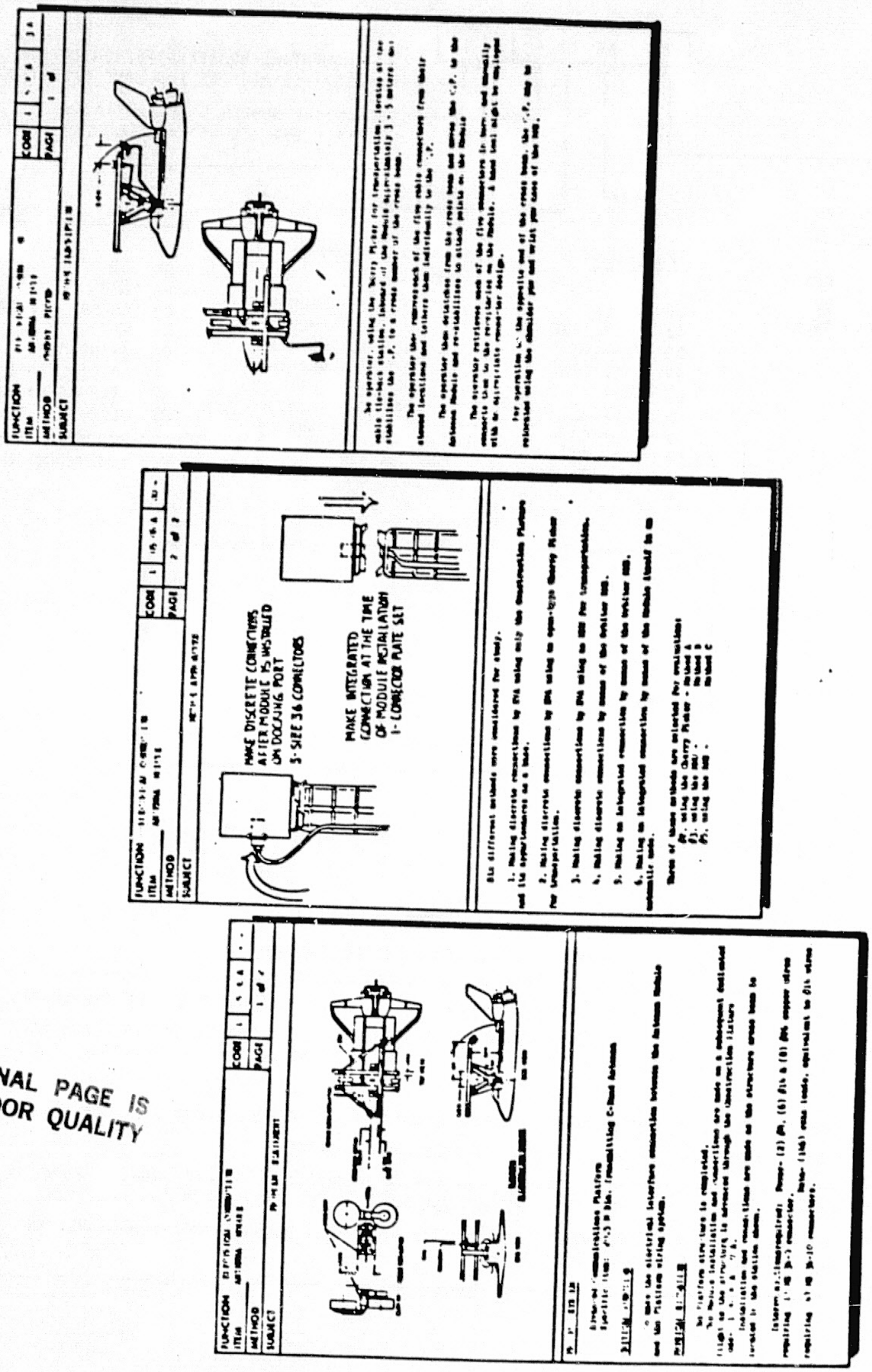
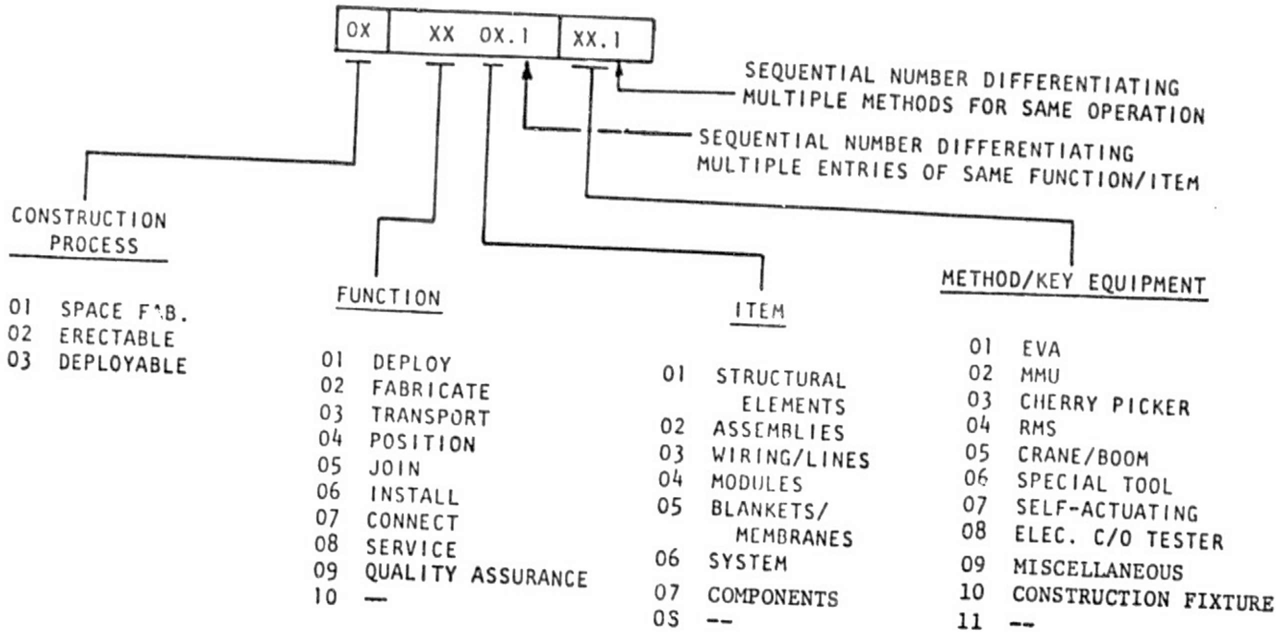


Figure 2.2-17. Sample Data Base Pages



EXAMPLE:

01 03 06.1 04

CONSTRUCTION PROCESS: SPACE FABRICATED
FUNCTION: TRANSPORT
ITEM: SYSTEM (FIRST ENTRY IN DATA BASE FOR "TRANSPORT SYSTEM")
METHOD: RMS (FIRST METHOD FOR THIS OPERATION USING RMS)

Figure 2.3-1. Construction Data Base Code Explanation

Table 2.3-1. Item Definitions

01	STRUCTURAL ELEMENTS - INDIVIDUAL PIECES USED TO FABRICATE STRUCTURE OF THE SPACECRAFT.
02	ASSEMBLIES - AN ITEM WHICH IS COMPRISED OF SEVERAL STRUCTURAL ELEMENTS WHICH HAVE BEEN ASSEMBLED ON THE GROUND OR ON ORBIT BUT PRIOR TO BEING JOINED TO THE BASIC STRUCTURE.
03	WIRING/LINES - ELECTRICAL OR FLUID LINES.
04	MODULE - END ITEM REPRESENTING A MAJOR SUBSYSTEM OR PAYLOAD ELEMENT OF THE PLATFORM.
05	BLANKETS/MEMBRANES - LONG, NARROW, AND/OR THIN SURFACES.
06	SYSTEM - A PACKAGE SIMILAR TO A MODULE DURING TRANSPORT TO ORBIT AND INSTALLATION ON THE BASIC STRUCTURE, BUT ONE WHICH IS UNFOLDED OR DEPLOYED AFTER INSTALLATION.
07	COMPONENT - A PART (INSTRUMENT OR BRACKET) WHICH MAY BE USED INTERCHANGEABLY IN MULTIPLE APPLICATIONS ON THE PLATFORM.

The individual method descriptions contain several pages of general information pertinent to each of the methods. These data include the project the data were based upon, a simple statement of the operation, the physical situation, and a list of all the methods identified. The physical situation delineates the condition of the project at the start of the operation being covered, and the ground rules and assumptions as applicable. The physical situation is meant to clearly identify a common starting point for each of the methods so that a true comparison of the methods can be made by the user. The basic format for each of the methods includes a specific page or pages, as applicable, for the following subjects.

1. Method Description

2. Project Modifications—Changes to the project configuration which are peculiar to the method being discussed.

3. Operations—In addition to the manpower requirements and estimated time to perform the actual operation, the "Supporting Activity" is also identified. This is used in most cases to identify the time to perform tasks which are pertinent to the operation being described but are of a one-time nature, and thus are not included in the activity time for a repetitious type of operation.

4. Construction Support Equipment Requirements—The basic construction fixture has not been included as it is common to all methods for a particular construction project.

5. Support Services—The support services are those to be provided by the construction base; in this case, the orbiter. The electrical requirements for the basic operation of the fixture (welding, translation, etc.) and the beam machine have not been included, as these requirements can only be determined from an integrated construction analysis. Two numbers are shown for the crew requirements: the one on the left (top) is the number of different individuals, and the one on the right (bottom) the average usage of the individuals to perform the operation. The operations time is that required to perform the generic operation. For example, even though there are 16 antennas to be installed on the communications platform, the time shown is only to install one. Thus, the data are more representative for other similar antenna installations.

6. Summary—The data presented on these pages are of the same nature as that described above for the Support Services pages.

In some cases, additional pages have been included to provide a more complete package on a particular method.

2.3.3 Construction Support Equipment (Section III)

This section includes general information regarding the major pieces of common construction support equipment that were used to support the various construction methods. These include the RMS, MMU, space fabrication beam builder, and the manned remote work station (cherry picker). The latest available reference material was used in the descriptions of these important construction equipment items.

2.3.4 Indexes (Section IV)

This section contains three indexes: (1) Function, (2) Item, and (3) Methods/Key Equipment. These titles refer to the major headings associated with the method code. The indexes are included to provide additional means of entering the data base. Thus, should a user of the data base be interested in methods associated with installation, he can look in the Function Index under "06 Install" and find nine operations, each of which includes two to three methods.

3.0 CONSTRUCTION SUPPORT SERVICES

The support services analyses presented here are a complementary set of investigations necessary to the development of alternative construction methods. Support services complement the basic construction materials and equipment to aid in the determination of the total mission resources required for space construction of given project systems. They include attitude control and stabilization, thermal control, illumination and TV, electrical power, and other necessary supportive functions. Initially, as presented here, they are focused on basic construction requirements issues. Later, during Part 2 of the study, they will be directed toward integration of the construction process where profiles of support services usages will be used in the evaluation and selection of preferred construction methods and processes.

Four important support services issues are presented. The nature of the attitude control problem during space construction is discussed including disturbance effects introduced by continually changing mass properties during construction. Also treated under the attitude control subject is the revisit and berthing problem. A preliminary safe closure criterion is presented along with concepts for meeting this criterion.

The potential need for thermal control of sensitive elements during the construction process, after installation but before system activation, is discussed.

The results of preliminary looks at the visibility/illumination problem during space construction are presented. Visual interference considerations from the sun and other bright sources are included along with electrical power illumination interrelationships.

Preliminary electrical power estimates for various individual elements in the construction process are presented along with their implications on the integrated construction process.

3.1 ATTITUDE CONTROL DURING CONSTRUCTION

Construction in space of large spacecraft will result in large changes in mass distribution and configuration. Thus, the attitude control system design must meet a variety of conditions. The following discussion reviews potential control requirements and control system alternatives leading to the selection of an acceptable control method for space construction operation. Then a construction scenario, based on the space fabrication of an advanced communication platform, is reviewed in terms of the selected attitude control method with detailed analysis of the salient features of the process.

3.1.1 Control Requirements and Approaches

Spacecraft control requirements during construction arise from the typical pointing and stability constraints of operational spacecraft and the special constraints resulting from the construction process. These are discussed in other sections of this report and are summarized on Table 3.1-1.

From these control requirements, a number of attitude control concepts may be employed. The relative merits and disadvantages of these approaches are presented below.

- Spin Stabilization—Induces larger loads than the other options and is not appropriate during docking and construction. It is rejected on this basis.

- Gravity-Gradient Stabilization—Attractive in that it provides a relatively quiescent, disturbance-free environment for delicate construction operations and partially built weak structure. However, construction and environmental disturbances can induce librations, thereby posing a possible requirement for libration damping.

- Orbiter Reaction Control System (RCS)—Attractive for libration damping and active orientation to stable gravity-gradient orientations after orbiter docking and prior to undocking. The vernier RCS does not have torque couples and loses control authority when the system mass center of gravity moves significantly outside the payload bay. The primary RCS can continue to provide control when the structure can withstand the loads induced by the larger thrust of this system.

- Operational Spacecraft Systems—These systems may provide libration damping and complete control after installation; however, their installation and the availability of power may not be feasible until late in the construction process.

- Dedicated Control Systems for Construction Only—It is desirable to eliminate the additional mass and cost of this class of system when possible. However, simple modular add-on systems for libration damping may be required and are relatively simple.

The gravity-gradient stabilization approach appears to be capable of meeting the construction requirements and has a minimal requirement for additional control equipment. It provides a relatively disturbance-free environment and is suitable for all phases of construction of the example configuration. It is selected on this basis.

3.1.2 Gravity-Gradient Stabilization During Construction

For satellites in circular orbits and those that rely on gravity-gradient stabilization, the principal axes of inertia must be aligned with the radial, tangential, and normal axes of the orbit in order for the satellite to maintain

Table 3.1-1. Attitude Control Requirements

SOURCE OF REQUIREMENT	REMARKS
ANTENNA POINTING	• CONTINUOUS, HIGH DATA RATE COMMUNICATIONS ARE NOT MANDATORY
THERMAL	• OPEN FRAMEWORK PRECLUDES EXCESSIVE SHADOWING TYPICAL STRUCTURE MATERIAL PROVIDES SUFFICIENT HEAT SHIELD
SOLAR ARRAY POINTING	• SPACECRAFT ATTITUDE AND RATE LIMITATIONS DEPEND ON SOLAR ARRAY GIMBAL AND GIMBAL DRIVE DESIGNS
ILLUMINATION	• SUN SHADES, FILTERS, AND PROPER SURFACE FINISHES REDUCE POINTING REQUIREMENTS IN SUNLIGHT • ARTIFICIAL LIGHTING AND LOW-LIGHT-LEVEL TV CAN BE USED IN SHADOW
LOADS	• LARGE MOMENTS ARE UNDESIRABLE DURING DELICATE CONSTRUCTION OPERATIONS • SPACECRAFT LIBRATIONS CAN INDUCE ADDITIONAL LOADS
DOCKING	• DOCKING METHODS FOR LARGE SPACECRAFT ARE NOT RIGOROUSLY DEFINED • REQUIREMENT EXISTS TO NULL ATTITUDE RATES AT END OF CONSTRUCTION TO FACILITATE ORBITER REVISIT FOR SUCCEEDING PHASE OF CONSTRUCTION
DISTURBANCE TORQUES	• NOMINAL ATTITUDE AT OTHER THAN DISTURBANCE TORQUE NULL POSITION CAN PRODUCE RELATIVELY LARGE GRAVITY GRADIENT TORQUES WITH RESULTING LARGE PENALTIES IN CONTROL. THEREFORE NULL-TORQUE ORIENTATIONS ARE HIGHLY DESIRABLE.

a fixed attitude in the orbiting frame. The question of stability in this attitude was addressed by D. B. DeBra and R. H. Delp.[1] They developed the stability diagram shown on Figure 3.1-1. This diagram shows two regions for which a gravity-gradient satellite is stable: Liapunov stable in the Lagrange region and infinitisimally stable in the delp region.

The analysis leading to this diagram assumes that (1) the spacecraft is a rigid body with constant mass properties and geometry, (2) the only forces on the satellite result from and inverse-square gravity field, (3) the body is small enough that the attitude motions do not significantly affect orbital motion, (4) the orbit is circular, and (5) the attitude deviations from the equilibrium position are small. These conditions are satisfied during the space construction process only prior to and after the period of internal motion resulting from erection and fabrication of the system. During that period, Conditions 1, 2, and 5 are violated. However, the stability diagram remains useful in establishing the orientation at the start of construction and the allowable extent of construction in terms of changes in moments of inertia.

The salient events of space construction are now examined in terms of the gravity-gradient stability diagram. The construction scenario assumes that a 36,300-kg (80,000 lb), 200 m (660 ft) long space fabricated tri-beam structure is built from the Space Shuttle orbiter and five 4500-kg (10,000-lb) elements are added to the structure. These elements represent, for example, large communication antennas or large subsystem modules. These events, selected because they demonstrate the significant gravity-gradient problems, are listed below.

- Orbiter unattached to the structure. This occurs before construction starts or before docking or after undocking.
- Erection and positioning of the construction fixture.
- Initial fabrication of a tri-beam assembly.
- Move the structure through the construction fixture. This could be for the purpose of putting the fixture at the opposite end of the construction or adding relatively massless items along the length of the structure.
- Move the structure through the fixture and add five 4500-kg (10,000-lb) elements, equally spaced, to the structure

The stable gravity-gradient orientation of an unattached orbiter is nose down, or up, and wings parallel to the orbit plane. This orientation and the location of the inertial ratios $(I_y - I_x)/I_z$ and $(I_y - I_z)/I_x$ are shown on Figure 3.1-2. The erection and positioning the construction fixture starts

[1]DeBra, D. B., and R. H. Delp, *Rigid Body Attitude Stability and Natural Frequencies in a Circular Orbit*, Journal of the Astronautical Sciences, Volume 8 (January 1961) pp 14-17.

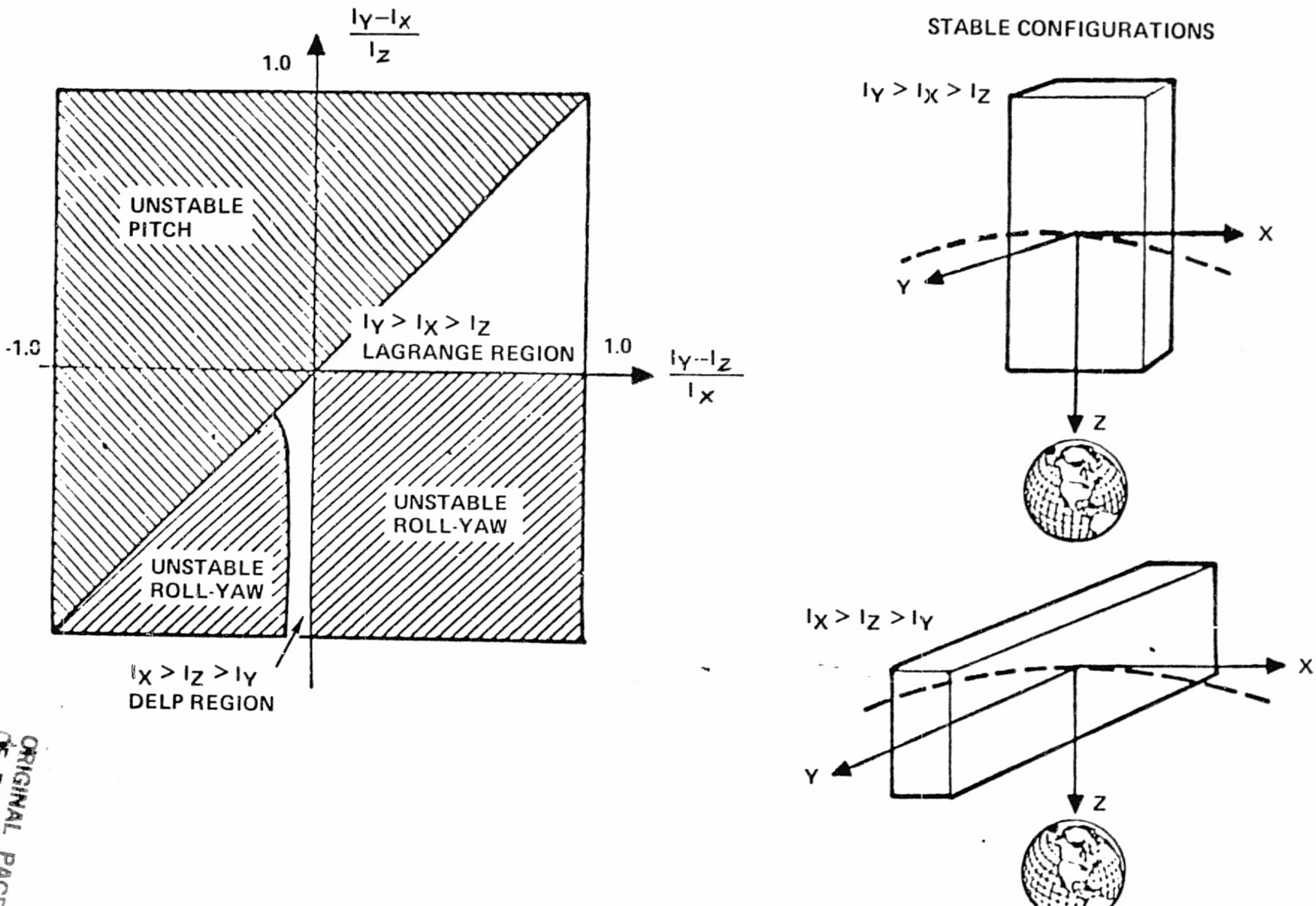


Figure 3.1-1. Gravity-Gradient Stability Diagram

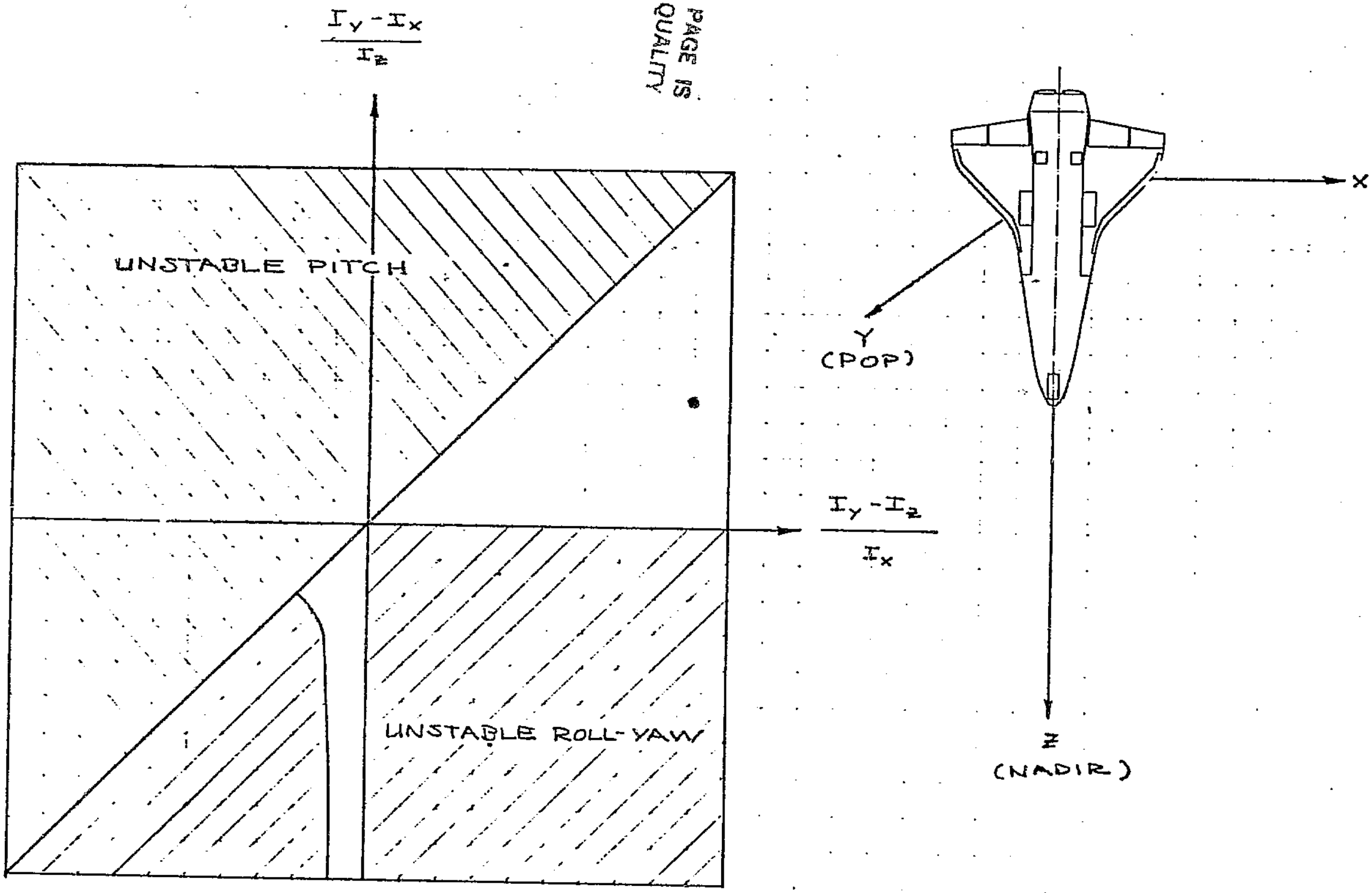


Figure 3.1-2. Gravity-Gradient Stability Diagram for Orbiter Only

with the orbiter in this orientation. The process is assumed to proceed in four steps as shown on Figure 3.1-3, which also shows the stability diagram for this operation. It can be seen that the orbiter-fixture combination becomes unstable because I_x, the orbiter's pitch moment of inertia, becomes larger than I_y, the orbiter's yaw moment of inertia. Attitude and rate histories are shown in Figures 3.1-4 and 3.1-5. Attitude, described in terms of Euler angles of a pitch, roll, yaw sequence and body rates diverge in roll and yaw as predicted by Figure 3.1-3.

The third case is the initial fabrication of a tri-beam assembly. Assuming that the angular rates are small enough, construction can proceed from the orientation at the end of the previously discussed event. If not, then active control would be required to damp the librations and maintain stability in this unstable region. An alternative would be to reorient the system into the stable region and then damp the residual rates to an acceptable level. In any case, the subsequent construction, the third event, would cause the inertia ratios to move into the third quadrant of the diagram. Although this is in an unstable region, slow changes in the relative magnitudes of the principal moments of inertia and their directions relative to body fixed coordinates will not excite large libration amplitudes. A second method of limiting these amplitudes is to provide active damping. This could come from operation of the orbiter reaction control systems in a damper mode or from the use of a simple and inexpensive add-on reaction control system.

The fourth case is the movement of the structure through the construction fixture while adding relatively massless items. Two cases were considered to determine the effect of speed of construction on attitude dynamics. For each of the cases, it is assumed that construction starts at the gravity-gradient stable orientation as shown on Figure 3.1-6. The loci of the inertia ratios are also shown on the figure.

The first case (Figures 3.1-7 through 3.1-10) is for a construction speed, or translation of the structure through the construction fixture, of one meter per minute. The second (Figures 3.1-11 through 3.1-13) is for a construction speed of about 1/3 meter per minute (1.0 ft/min.). The initial conditions for each are such that the principal axes of inertia are aligned with the radial, tangential, and normal axes of a circular orbit at an altitude of 300 nmi. The Euler angles describing the orientation of the body axes are $\theta = -3.556°$, $\phi = -0.6758°$, and $\psi = 16.18°$. The initial body rates are $p = 0.0175$ deg/sec, $q = 0.0602$ deg/sec, and $r = 0.0007$ deg/sec. This places the long axis of the structure parallel to the local vertical and the orbiter roll axis parallel to the velocity vector. Construction starts at 96 minutes in each case and ends at 296 minutes for the fast case and at 756 minutes for the slow case.

Figure 3.1-7 shows the elements of the inertia matrix as a function of time for fast construction. The change in the elements is a result of the translation of the structure through the construction fixture. The moment of inertia history for the slow case is the same, but occurs over the longer period of construction. A comparison of the Euler angle histories for the fast and slow cases is shown on Figures 3.1-8 and 3.1-11, respectively. The Euler angle histories for the fast construction case show that the spacecraft is rotating in yaw with unconstrained motion about the local vertical. The

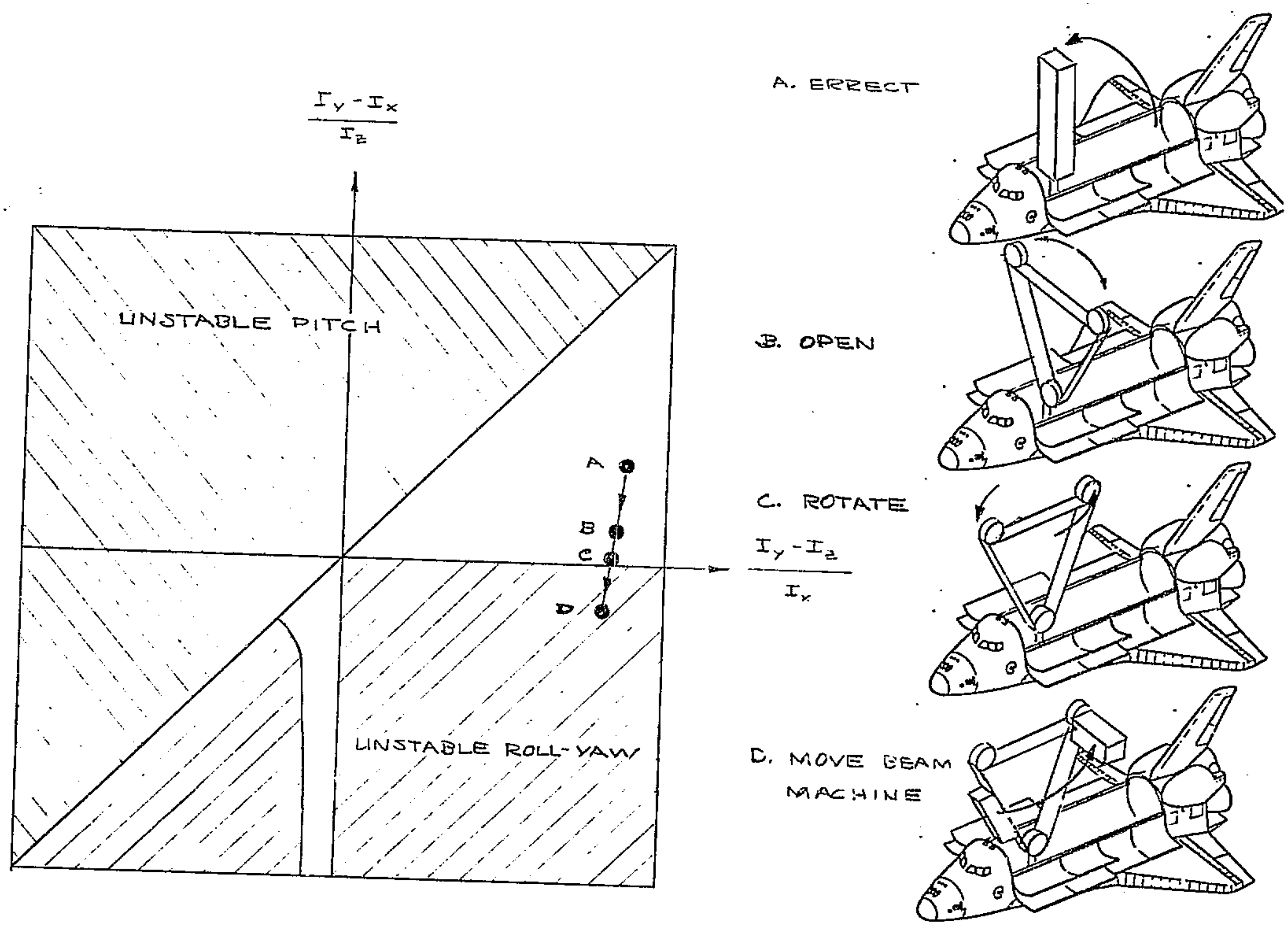


Figure 3.1-3. Gravity-Gradient Stability Diagram for Fixture Erection Case

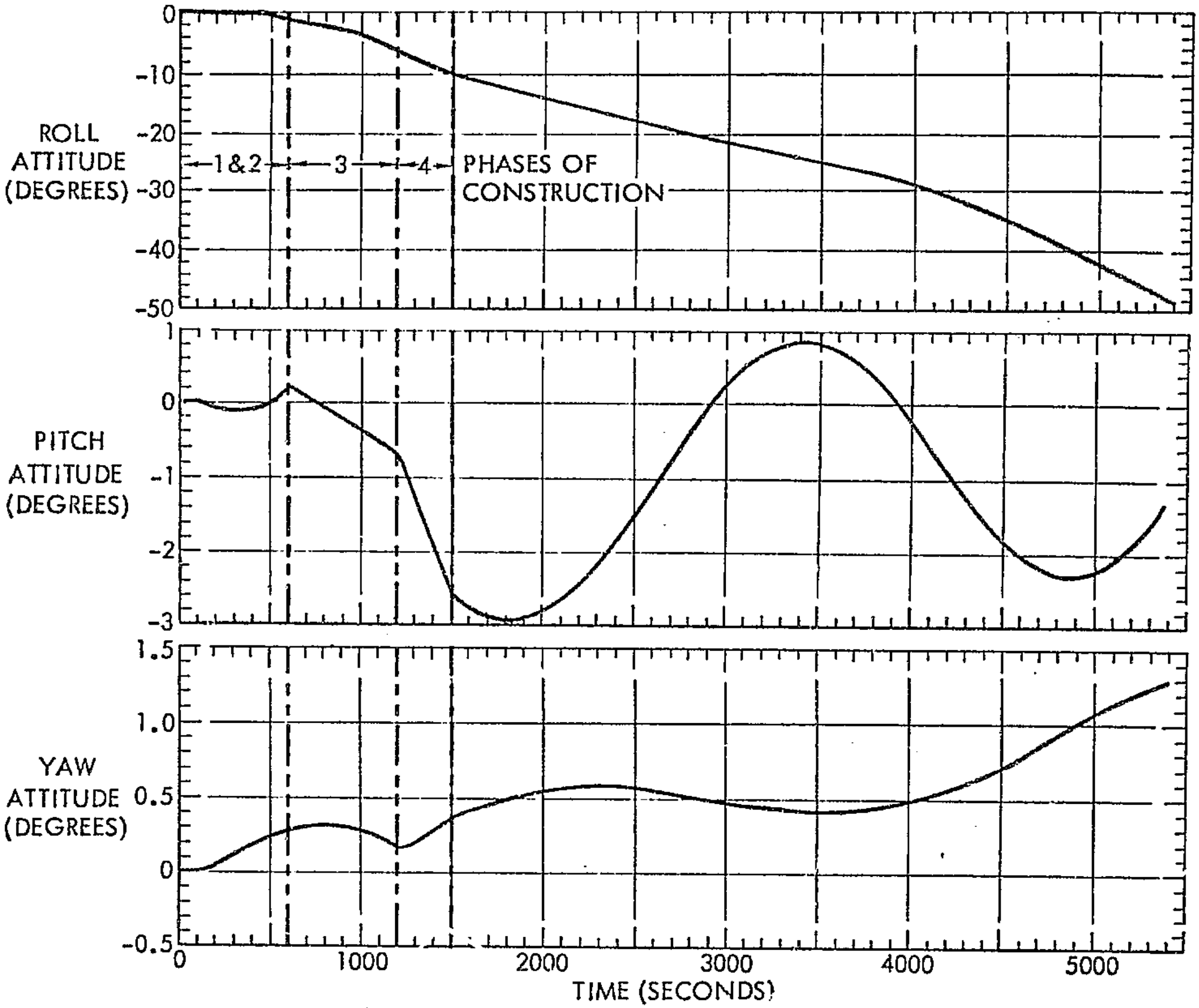


Figure 3.1-4. Attitude Histories for Fixture Erection Case

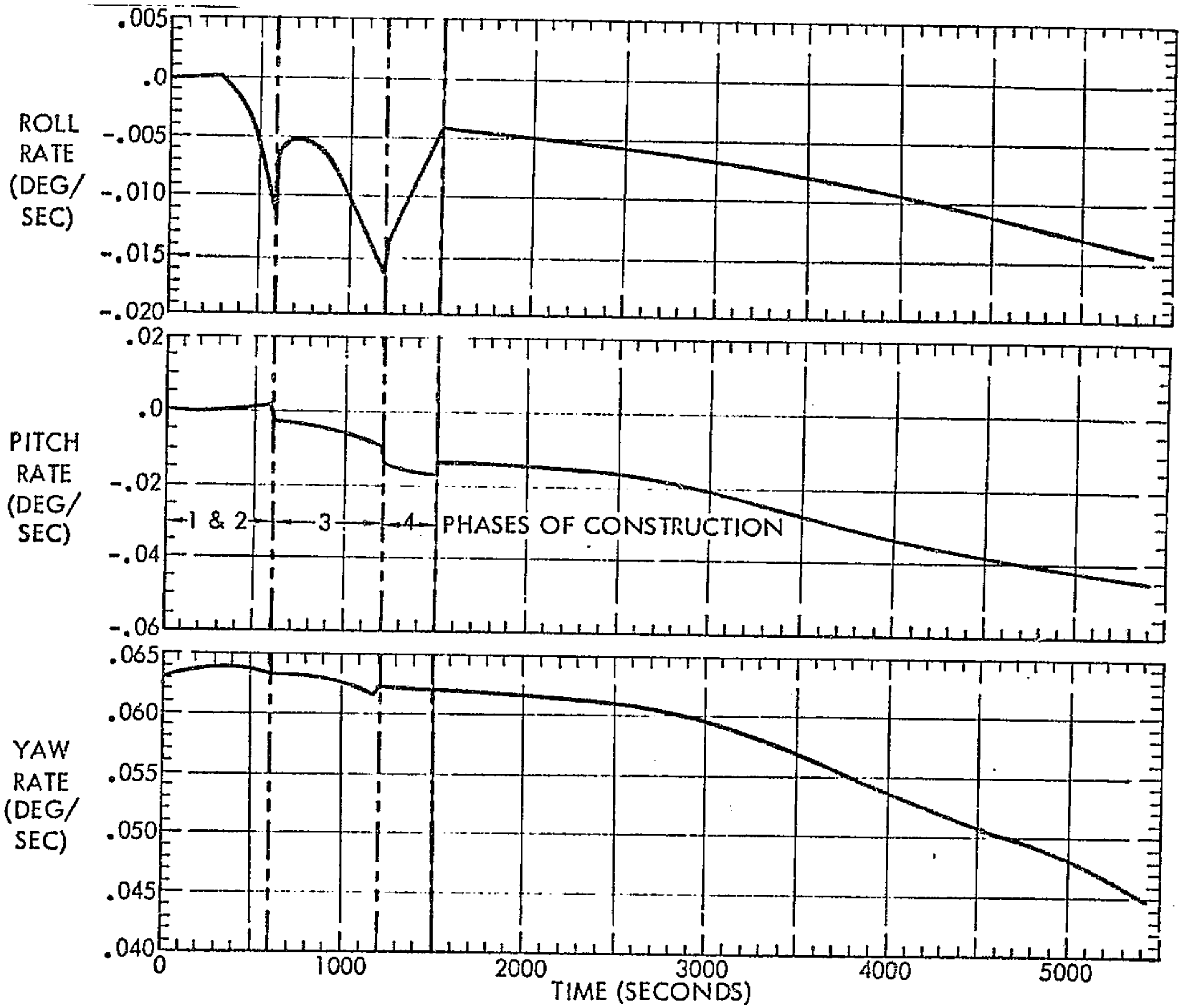


Figure 3.1-5. Body Rate Histories for Fixture Erection Case

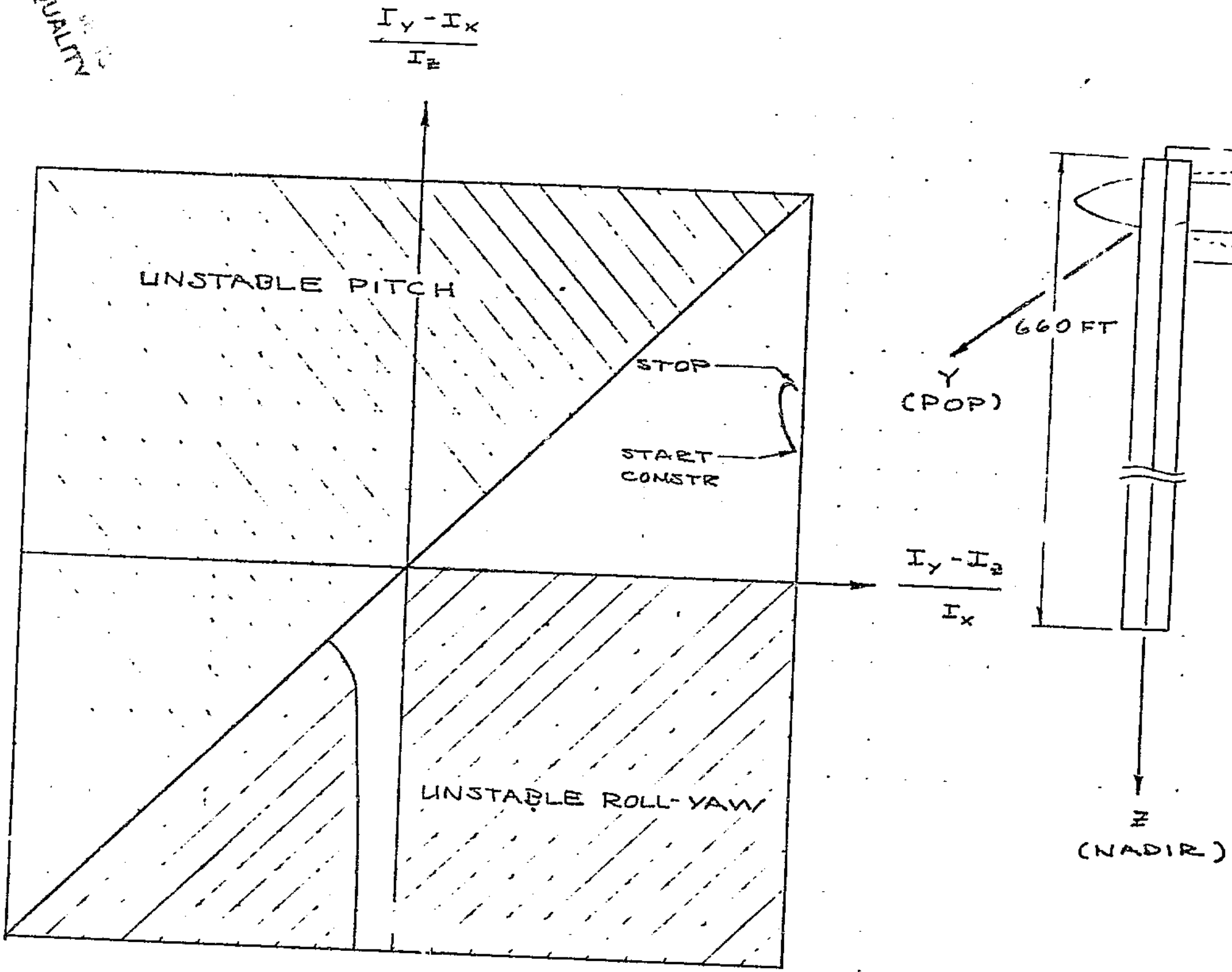


Figure 3.1-6. Gravity-Gradient Stability Diagram for Simple Translation of Structure

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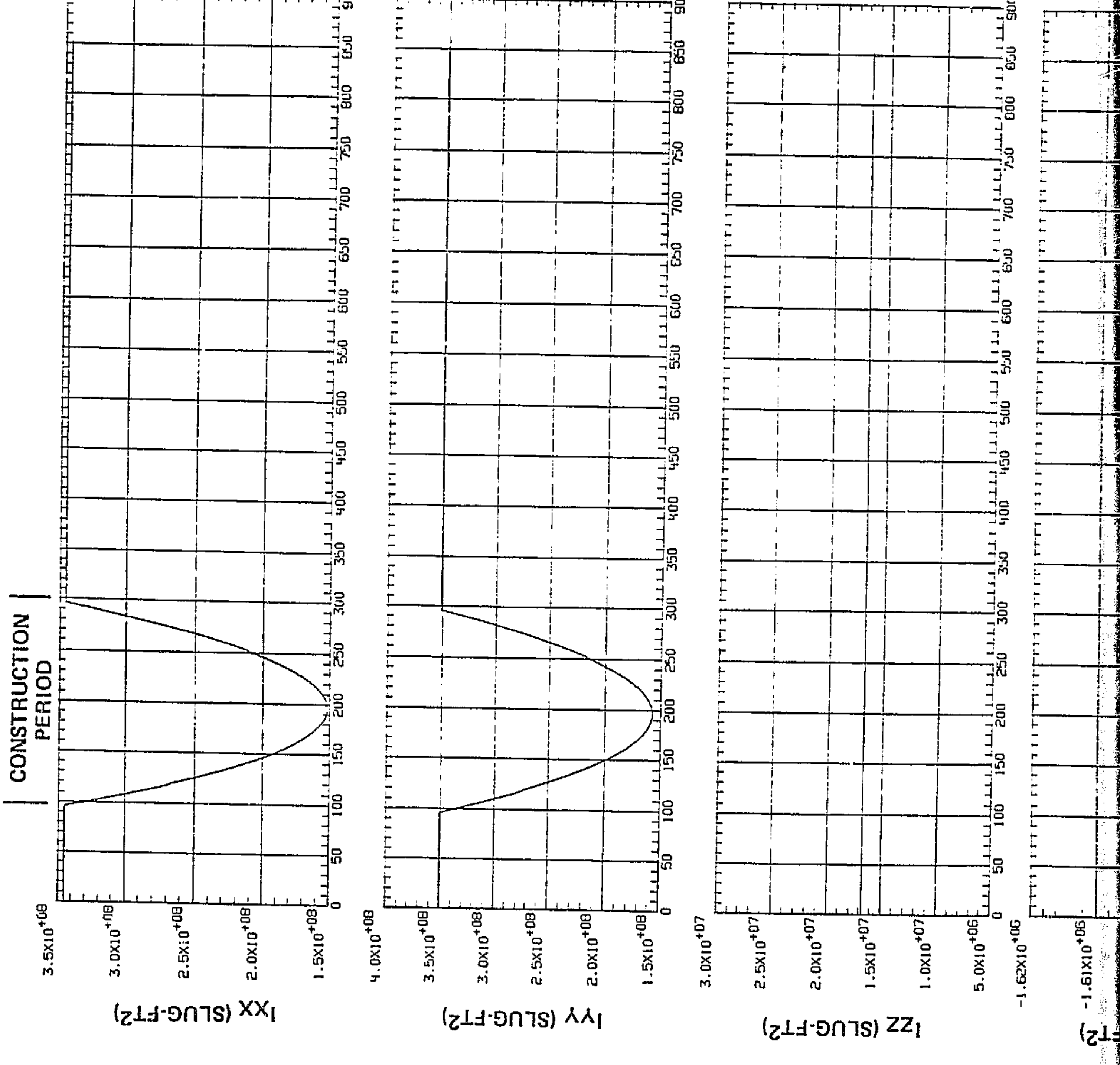

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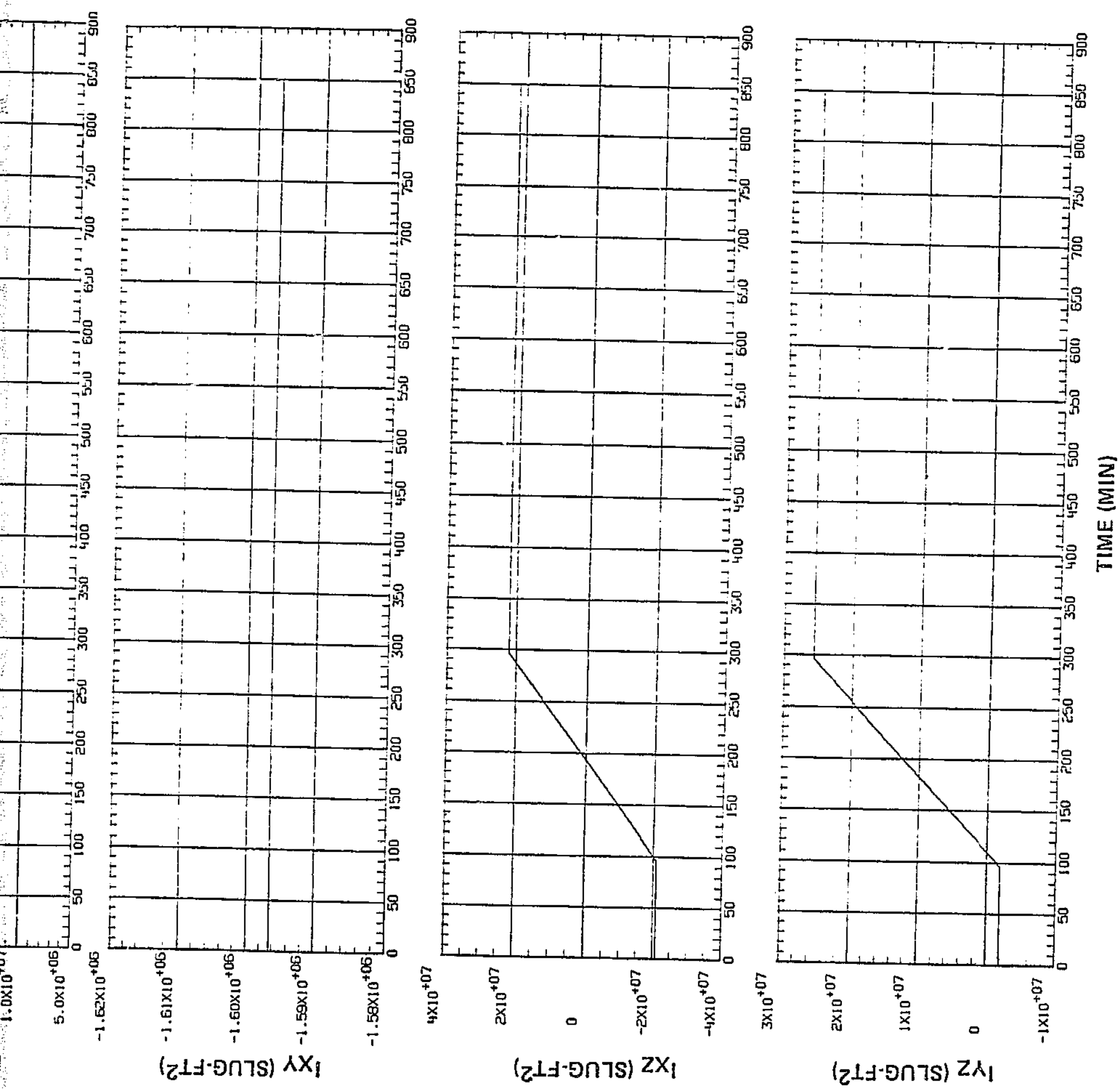

Figure 3.1-7. Moment of Inertia Histories for Fast Construction Case

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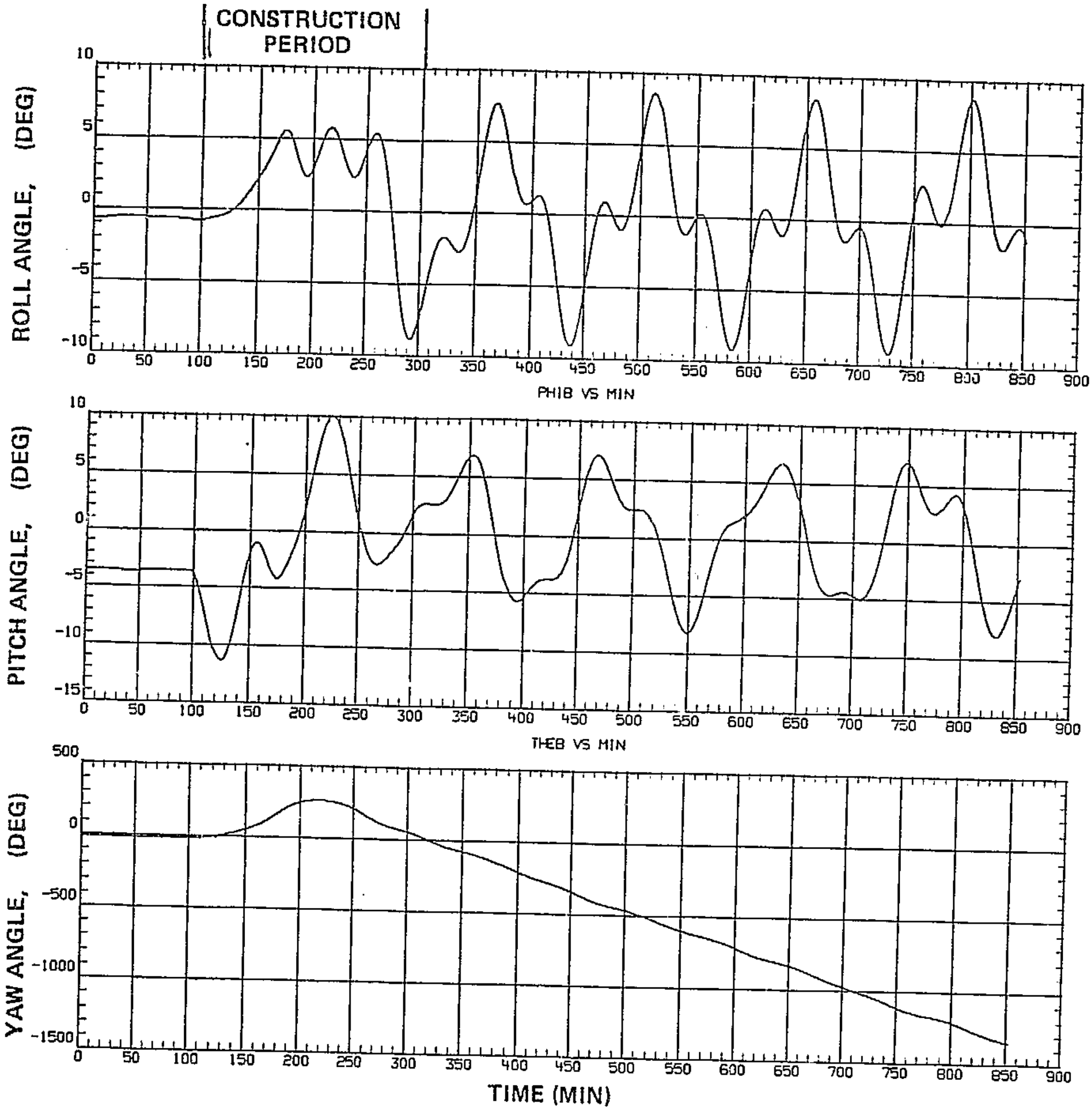


Figure 3.1-8. Euler Angle Histories for Fast Construction Case

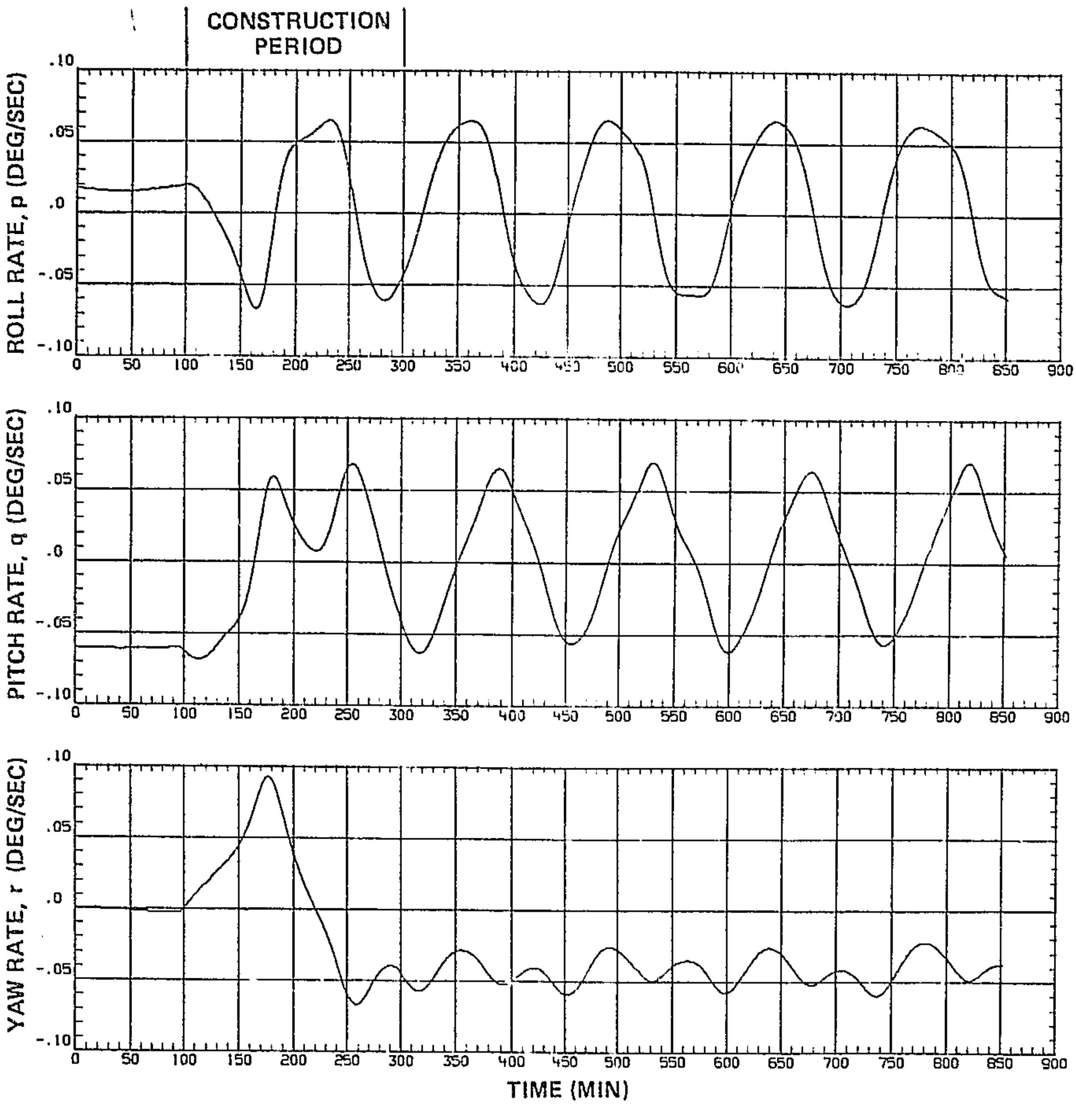


Figure 3.1-9. Body Rate Histories for Fast Construction Case

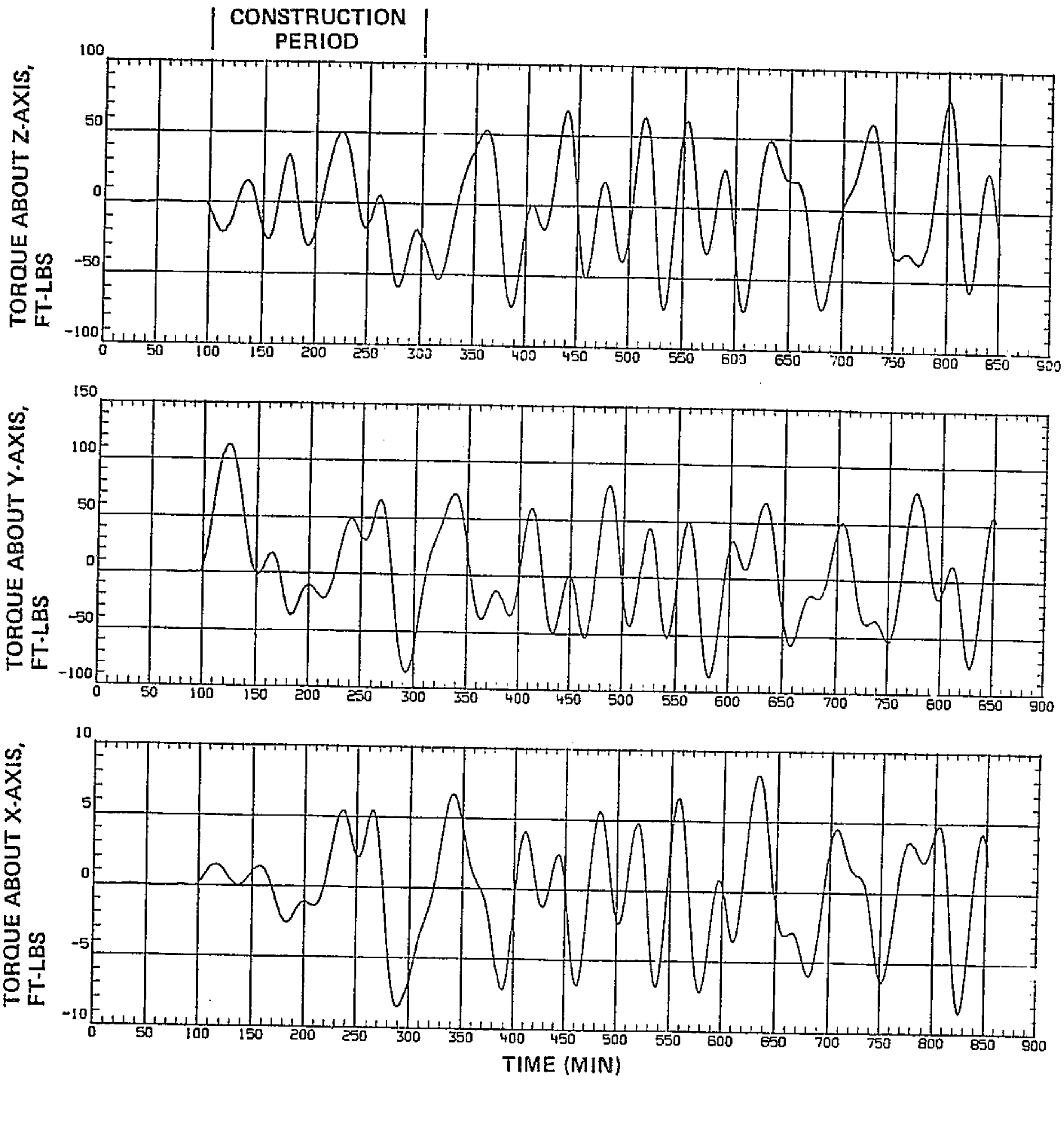


Figure 3.1-10. Gravity-Gradient Torque Histories for Fast Construction Case

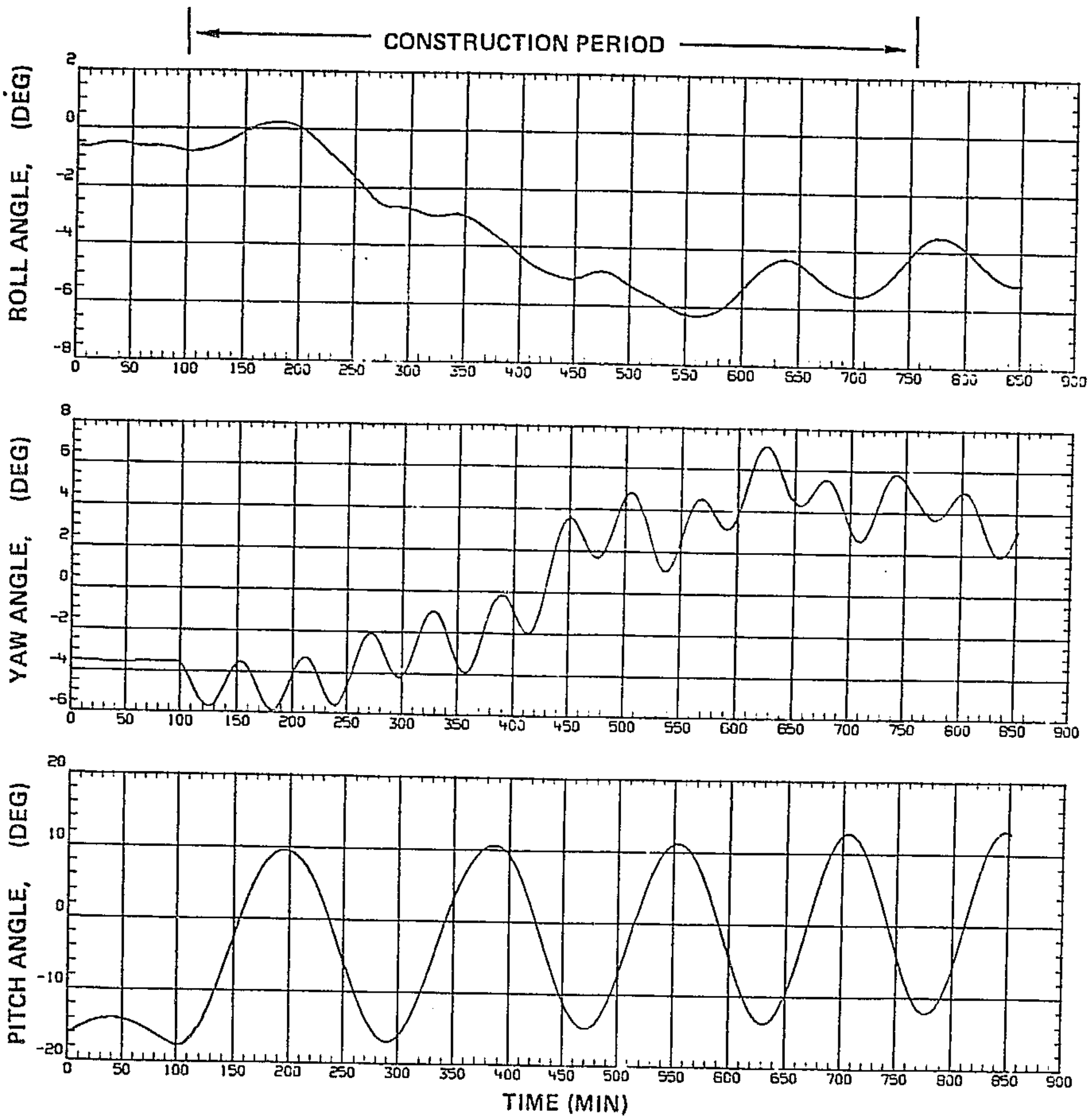


Figure 3.1-11. Euler Angle Histories for Slow Construction Case

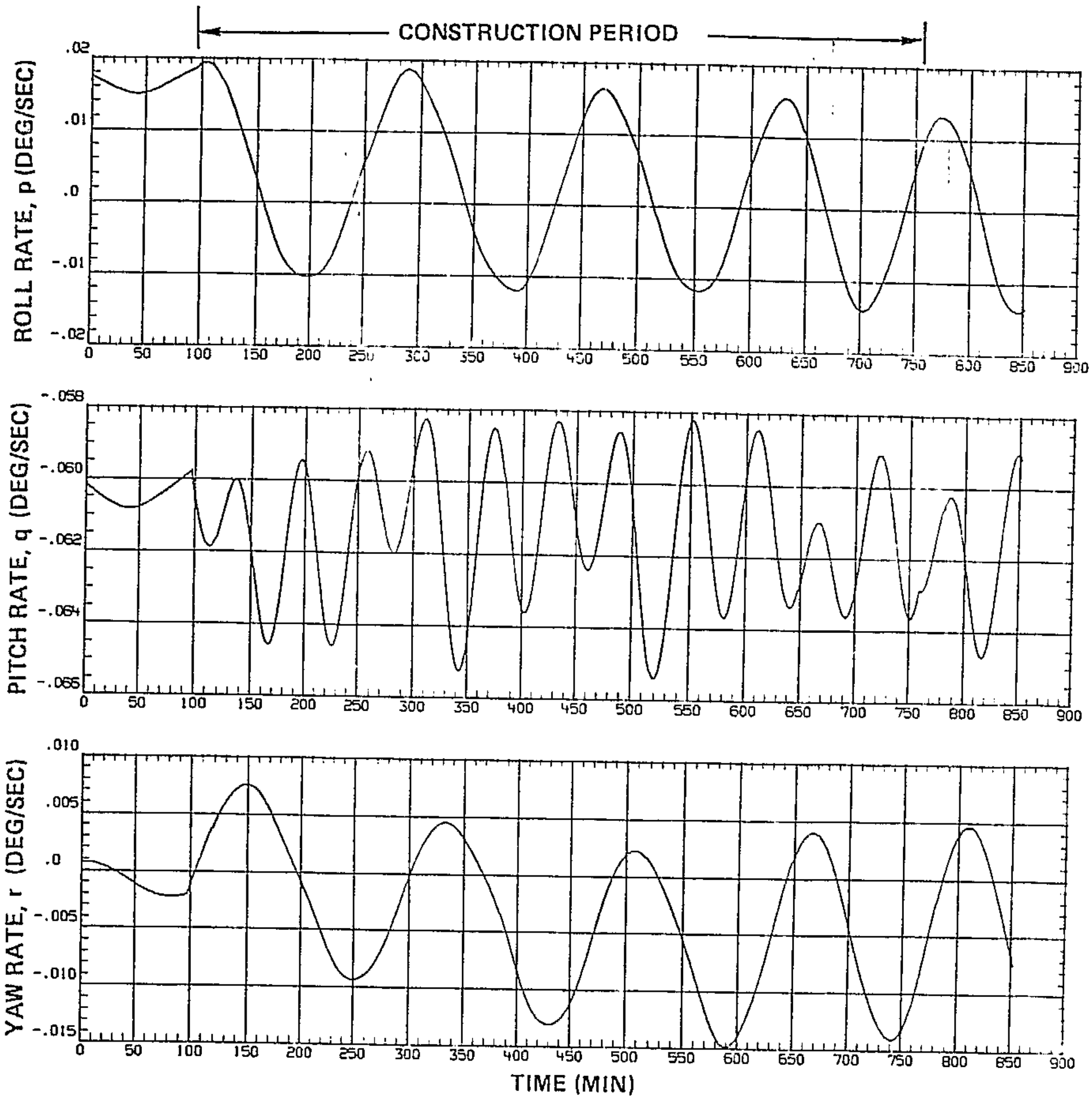


Figure 3.1-12. Body Rate Histories for Slow Construction Case

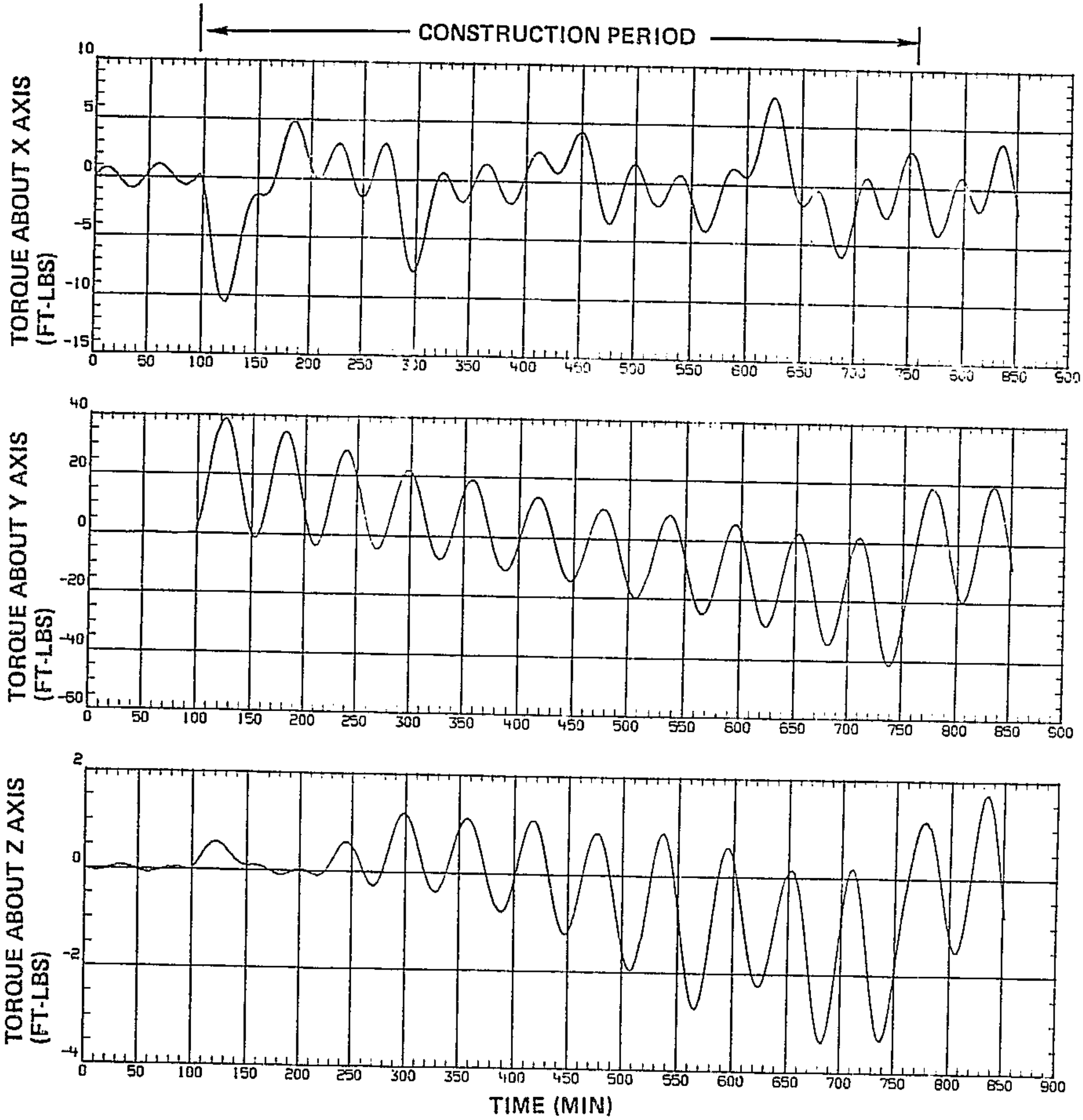


Figure 3.1-13. Gravity-Gradient Torque Histories for Slow Construction Case

use of the slower construction speed eliminates the tumbling and reduces pitch and roll librations. Hence, it is concluded that some degree of construction speed modulation can prevent tumbling and reduce the amplitude of libration and that use of gravity-gradient orientations during space construction will permit relatively long periods of uncontrolled, disturbance-free construction.

Figures 3.1-9 and 3.1-12 show the body rate histories for the fast and slow cases, respectively. Rates for the fast case are generally much larger than for the slow case, five times larger in roll rate and ten times larger in yaw rate. The gravity-gradient torques about the body axes are shown in Figures 3.1-10 and 3.1-13. The torques in the fast case are larger than in the slow case, which is expected because the attitude excursion and, hence, out-of-trim conditions are larger.

The last case considered is similar to the previous fast construction case with the exception that the translation motion is stopped periodically to add five 4500-kg (10,000-lb) elements to the structure. Figure 3.1-14 shows the construction scenario for this case, and Figure 3.1-15 shows the first quadrant of the stability diagram with plots of the inertia ratios for this case as well as a detail of Figure 3.1-6. The discontinuities in the plot are due to the movement of the elements from the payload bay to the structure. Figure 3.1-16 shows the inertia histories, and Figures 3.1-17 and 3.1-18 show the Euler angles and body rate histories. Figure 3.1-19 shows the gravity-gradient torque histories for this case. It can be seen that there is a large change in yaw attitude which is the axis with the weakest gravity-gradient stiffness. However, all body rates are small, not exceeding 0.07 deg/sec.

The results of the simulations of these several cases show that, for construction scenarios where large changes in mass distribution and dynamic changes in configuration exist, long periods of flight are possible without use of active control systems. A comparison of two identical cases with and without movement of masses from the cargo bay to the structure shows that there is an effect on the dynamics due to small inertia changes occurring periodically.

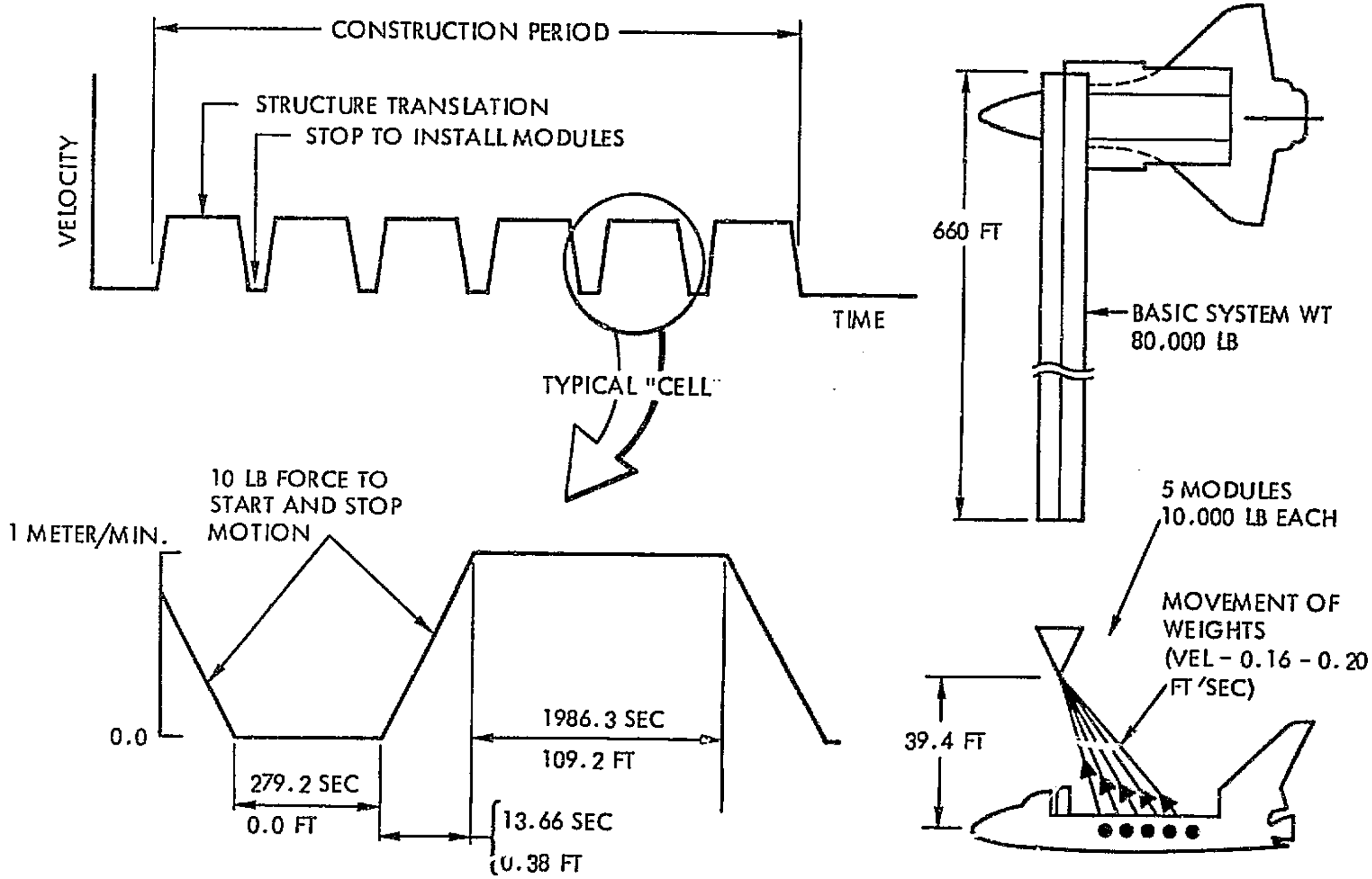


Figure 3.1-14. Construction Scenario for Added Weight Case

Satellite Systems Division
Space Systems Group
Rockwell International

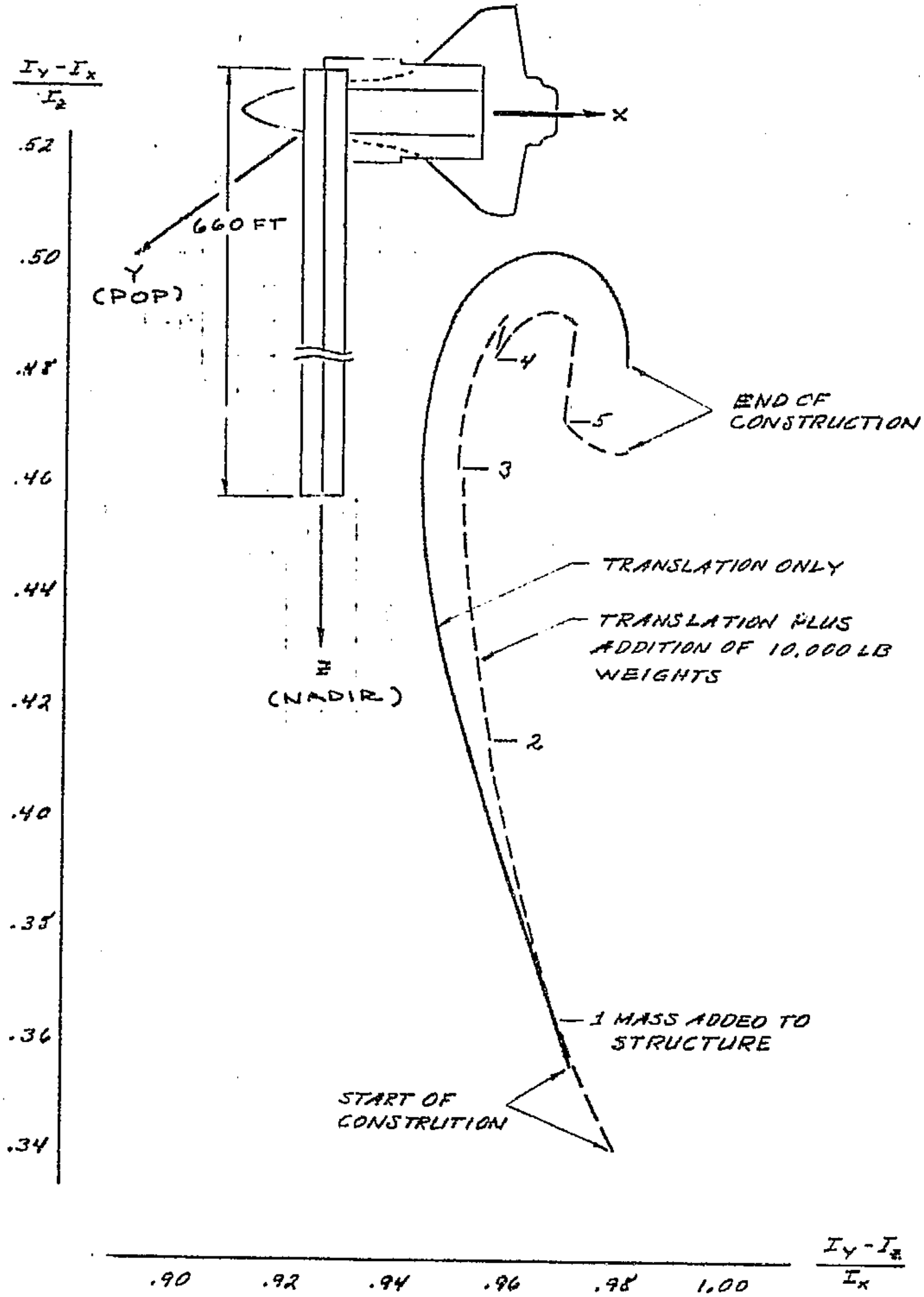


Figure 3.1-15. Stability Diagram Showing Construction in Construction-Stable Orientation

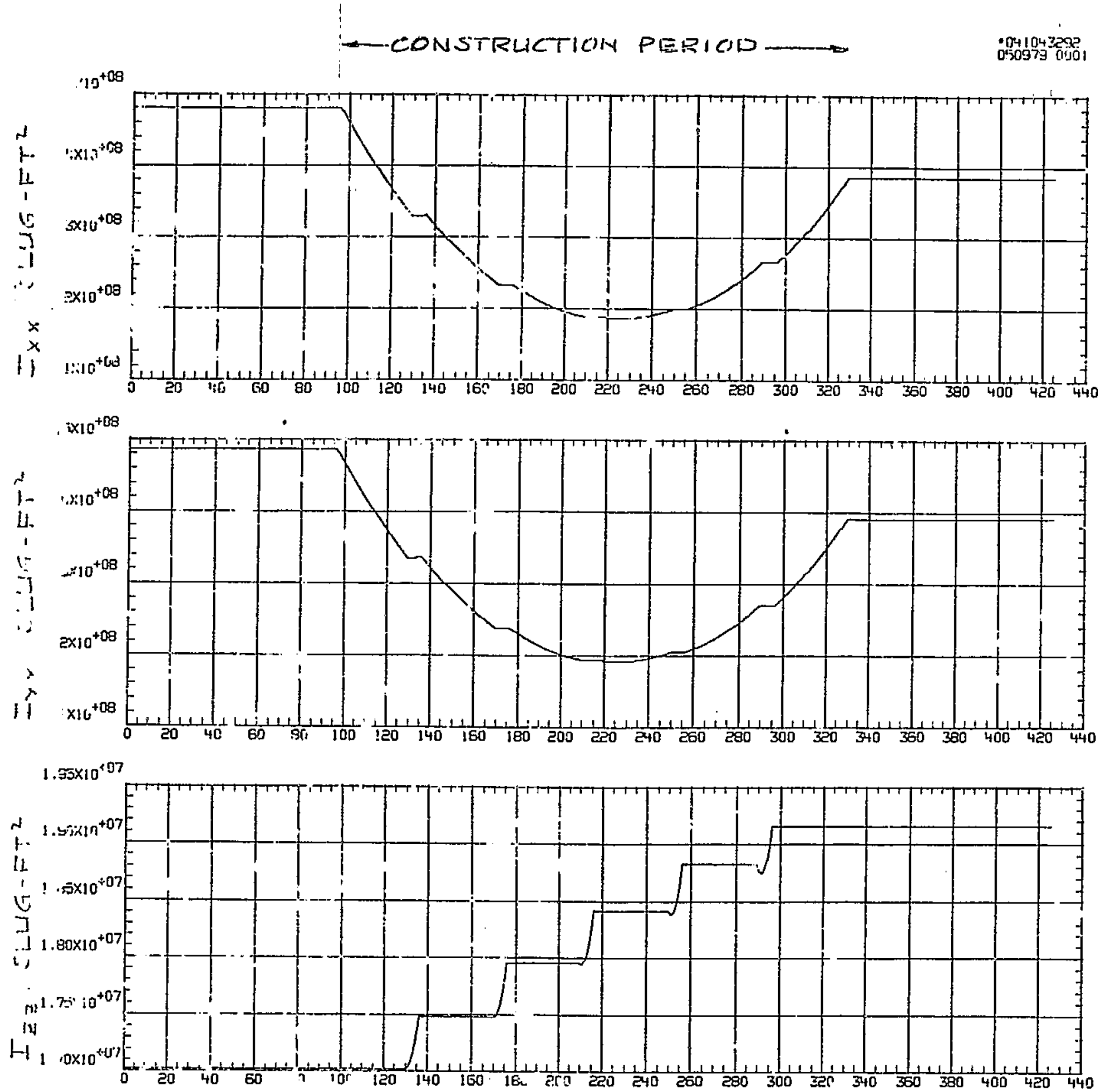


Figure 3.1-16. Moment of Inertia Histories for Added Weight Case

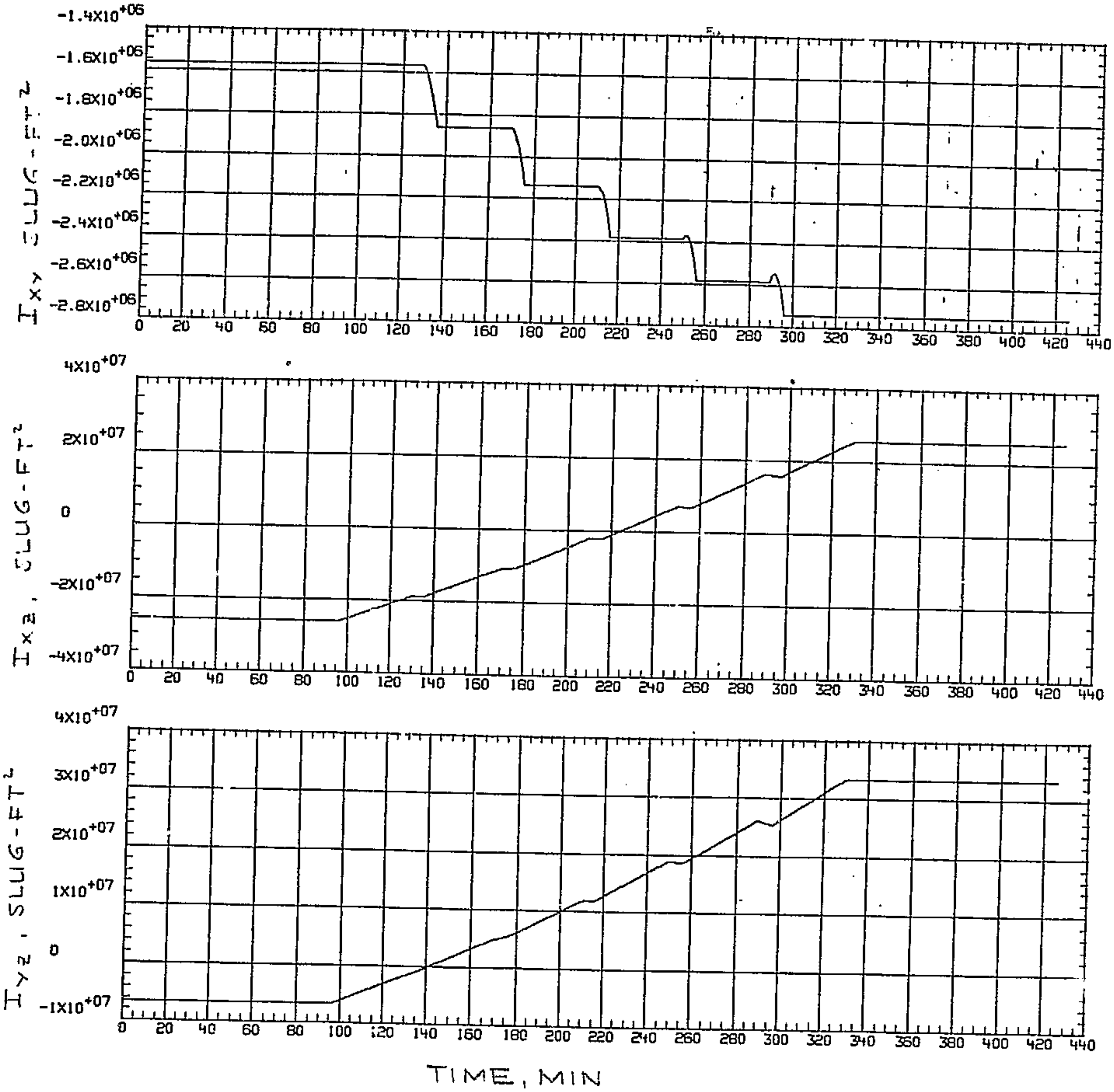


Figure 3.1-16. Moment of Inertia Histories for Added Weight Case (Cont.)

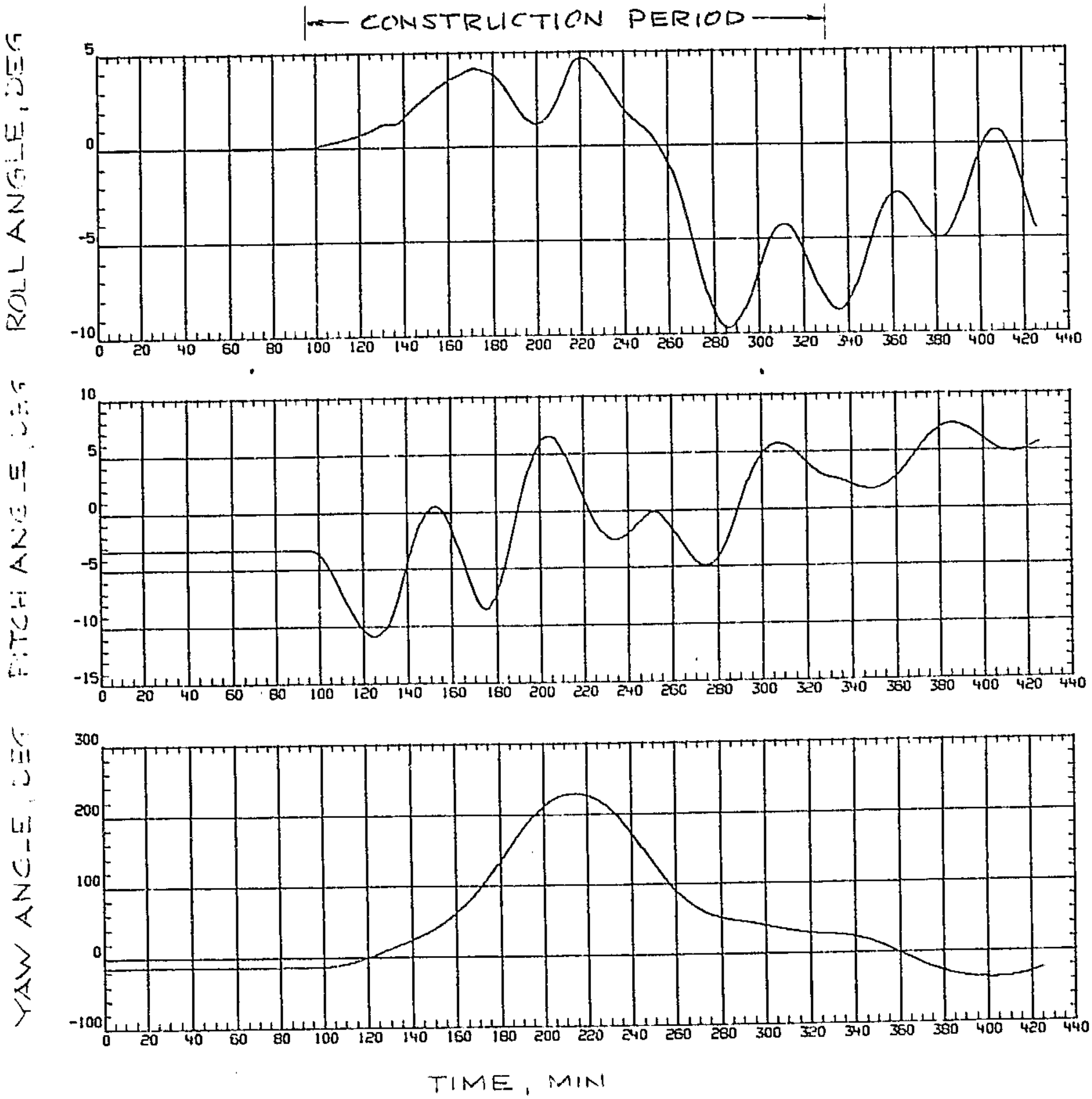


Figure 3.1-17 Euler Angle Histories for Added Weight Case

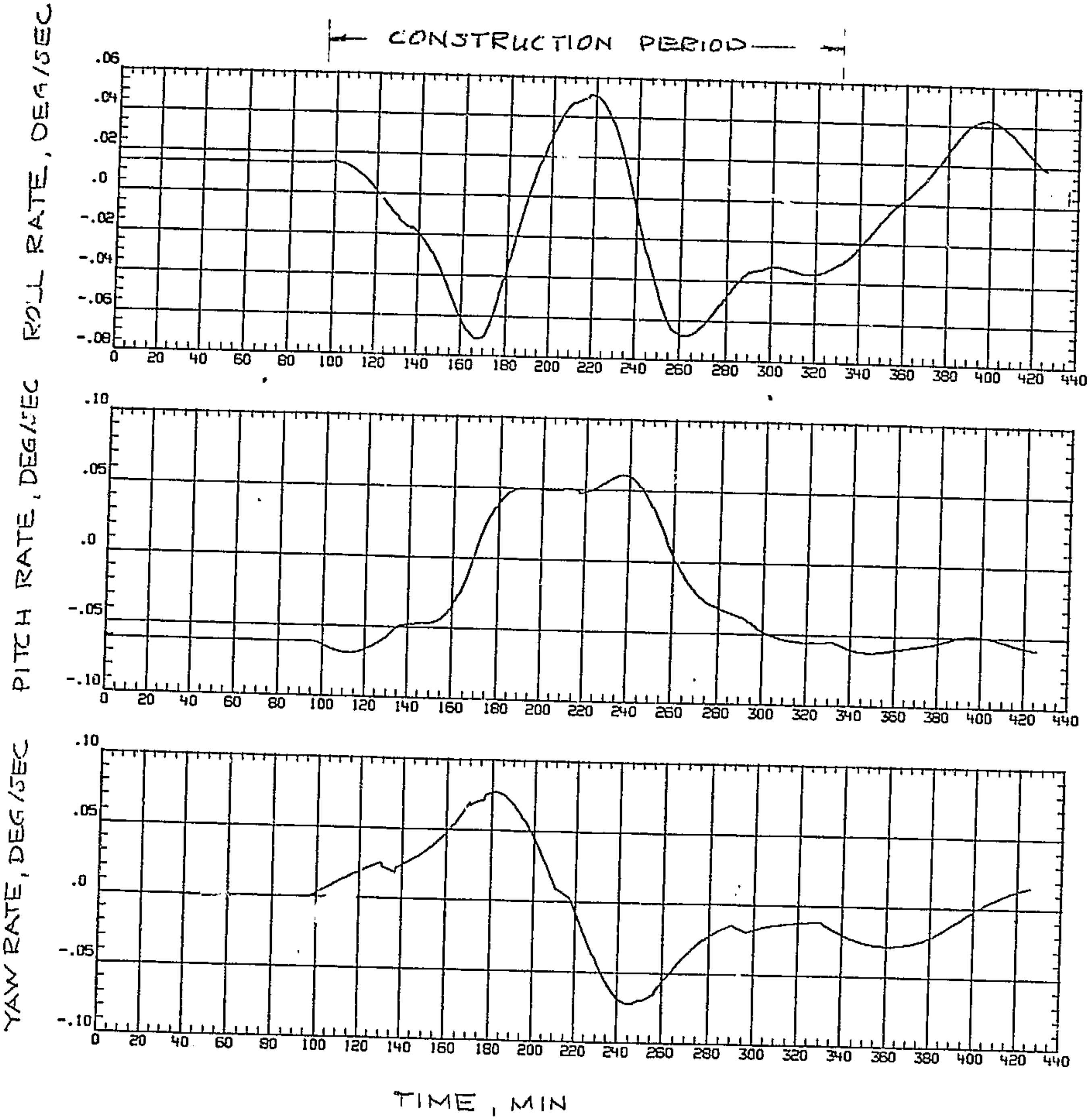


Figure 3.1-18. Body Rate Histories for Added Weight Case

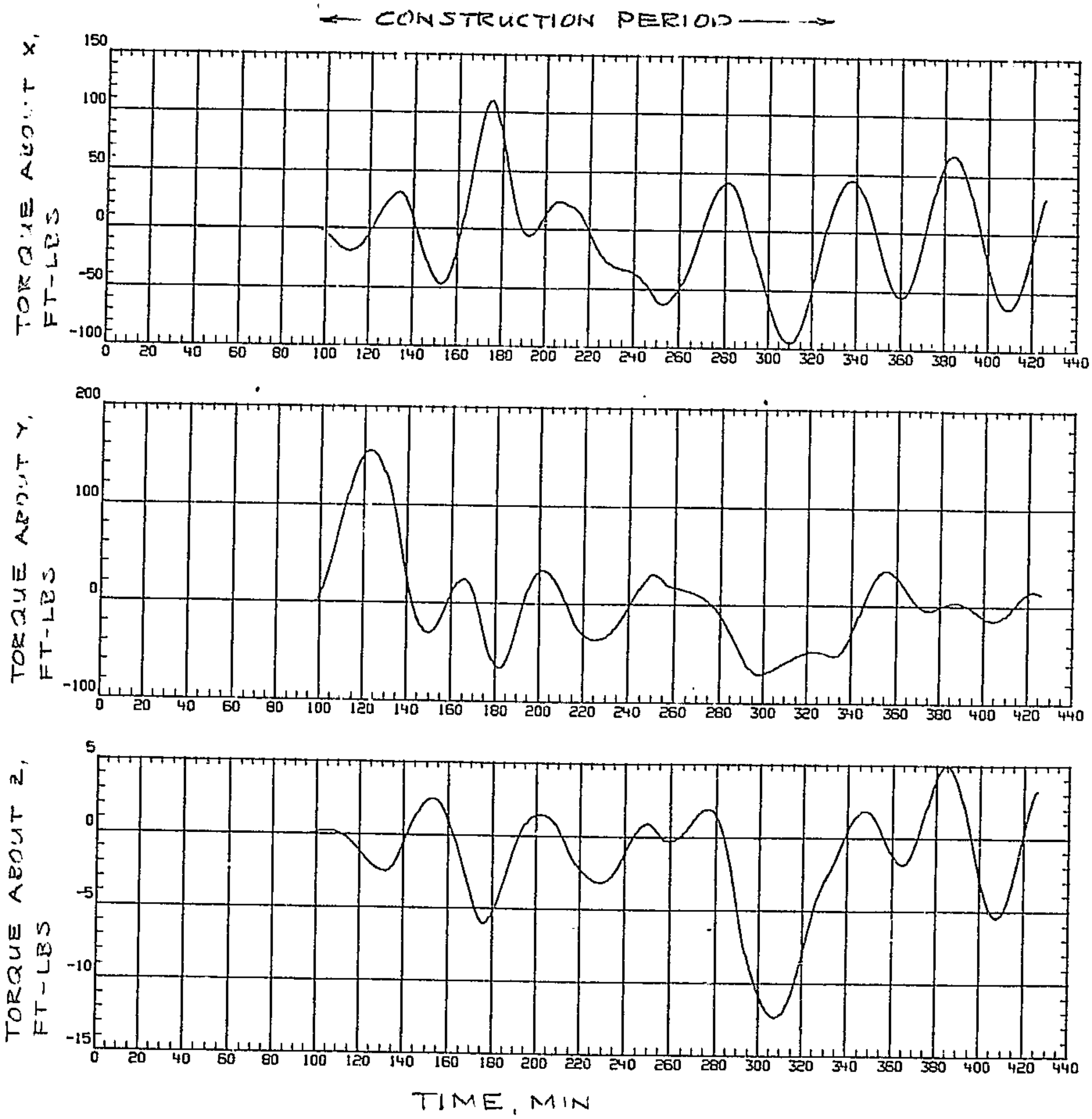


Figure 3.1-19. Gravity-Gradient Torque Histories for Added Weight Case

3.1.3 Revisit and Berthing

Large area space systems will typically require more than one Shuttle flight to complete the construction process. The project systems treated here require four or more flights. Thus, the capability for revisit and berthing to the partially completed space structure is a basic construction requirement.

The fundamental revisit "problem" is depicted in Figure 3.1-20. It is a two-body problem with each body having its own mass properties and six-degree-of-freedom (DOF) motion. The orbiter must be flown in a fail-safe approach trajectory that eliminates the possibility of an inadvertent collision with the target system. This requires precise ΔV control of the terminal closure path while simultaneously maintaining line-of-sight visibility to the target. This precision control is hampered by attitude control coupling with translation control due to RCS thruster geometry and minimum impulse size.

The target vehicle will typically be librating with amplitudes and rates that could be significantly affected by plume impingement from proximity RCS firings. The RMS must reach out, track, and engage the target structure and then arrest its relative motion. The capability of the orbiter/RMS combination to perform this sequence is greatly affected by the mass properties, hook-up geometry, and dynamic motion of the large area system to be engaged.

As a preliminary step in looking at this problem, a simplified model of the engagement dynamics was formulated. This model and the resulting safe engagement requirements are shown in Figure 3.1-21. In this model, the orbiter was assumed to represent an infinite mass, thereby reducing the situation to a one-body problem. The target mass properties and engagement geometry are shown in the figure. It was further assumed that the translation component of relative motion (X) must be arrested within 6 meters (20 feet). This allows ample time (100 to 200 seconds) for the RMS to track and grasp the target and then stop it before it travels out of reach. Similarly, the rotational motion was assumed to be arrested within a rotational angle of 30 degrees, thereby allowing adequate clearance between the orbiter and the target structure. Further, these "stopping" actions were constrained by the 67 newton (15-pound) tip force and 800 newton-meter (600 ft-lb) joint hinge moment limits associated with the RMS.

The resulting safe closure conditions are shown at the lower right of the figure. Relative $\dot{X}$'s range from 0.03 to 0.06 mps (0.1 to 0.2 fps) in combination with relative $\dot{\theta}$'s of 0.05 and 0.01 deg/sec. These are based on a single DOF model along with many other assumptions. Allowance for additional degrees of freedom will likely reduce these limits. Thus, careful attention must be focused on the librating motion of the partially completed space platform.

The librations of the platform when the orbiter returns for subsequent phases of construction are a function of the orbiter's ability to damp the motion prior to undocking plus the librations due to undocking disturbances. The orbiter has the capability to damp librations in either an attitude control limit cycle mode or a libration damping mode. The limit cycle amplitudes for an orientation such that the principal axes are parallel to the orbit tangential,

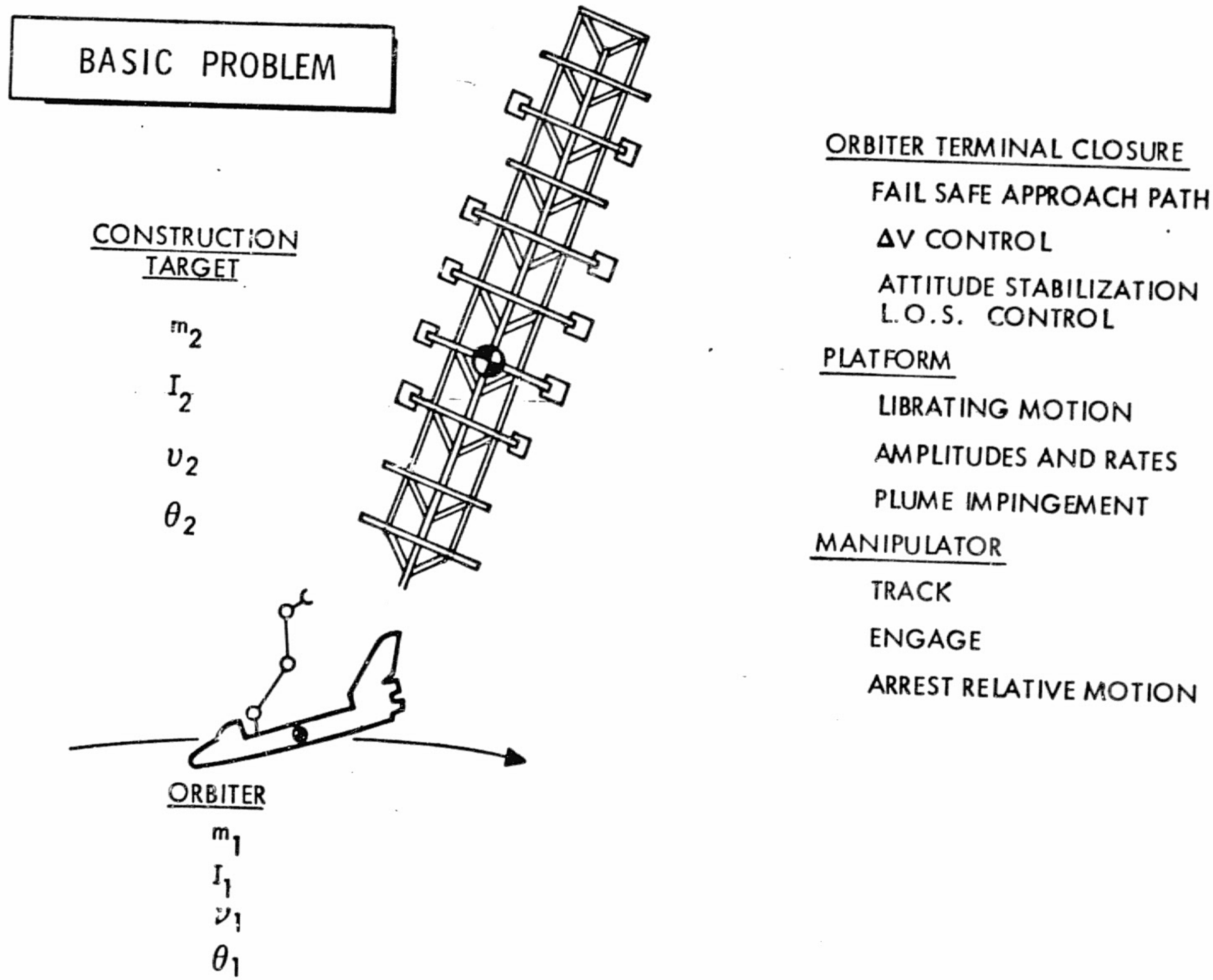


Figure 3.1-20. The Revisit/Berthing Problem

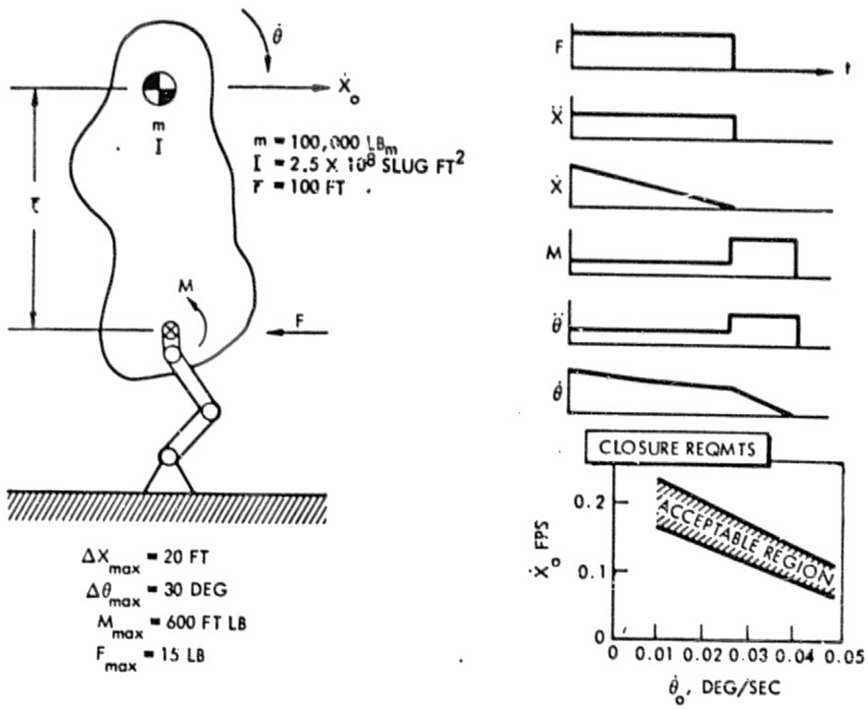


Figure 3.1-21. Safe Terminal Closure Criteria

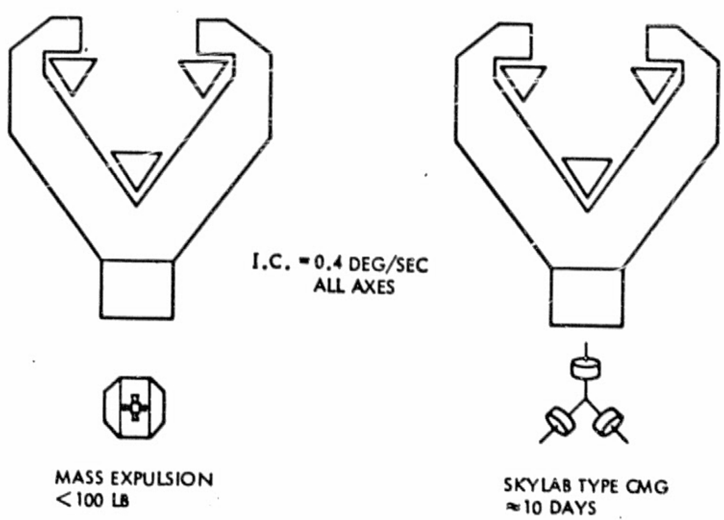


Figure 3.1-22. Libration Damping Concepts

normal, and radial axes are ±0.002 deg/sec in pitch and roll rate, ±0.0012 deg/sec in yaw rate, and ±0.1 deg in amplitude about each axis. However, these amplitudes will be different when the system mass properties are different from the orbiter-only situation. The combined orbiter/platform mass properties of the large area space systems studied here are greatly different from the orbiter alone and, thus, accurate projections of these limit-cycle capabilites and resulting motion are not presently available.

The additional librations induced by the undocking disturbance are a function of the unlatching loads, plume impingement, docking port location with respect to the system center of mass, and the platform orientation relative to its gravity-gradient null or "trim" orientation. These interactions require complex modeling to determine their magnitudes. Assuming they are negligibly small, then the motion limits which allow for undocking will meet the requirements for revisit. If the amplitudes are not negligible, then a libration damper in the platform/construction system would be required.

A libration damper could be contained in a module placed near the docking port for replacement or resupply as required. It must be lightweight and require minimum power such that the logistics would be a minor part of the orbiter's operations and cargo capability. A simple control system to operate the damper mechanism would be required. It would need to sense attitude and rate such that the librating motion would be damped sufficiently near the gravity-gradient null orientation. Two fundamental damping concepts for attenuating this motion, as shown schematically in Figure 3.1-22, are: a mass expulsion reaction control system and a momentum storage/exchange package.

A reaction control system placed at one end of the platform, on the construction fixture, would produce large pitch and roll torques for very low thrust levels. For a 36,000-kg (80,000-lb) tri-beam structure, 200 m (660 ft) long, and a libration rate about each axis of 0.4 deg/sec which is conservatively large, ten-pound hydrazine thrusters would null the motion in less than one day, using 44 pounds of propellant.

A relatively lightweight control moment gyro package such as the one used on Skylab would be sufficient. This weighs 170 kg (372 lb), including electronics, draws 320 W of power, and outputs 217 N-m (160 ft-lb) of torques. The momentum storage capability is 3100 N-m-sec (2286 ft-lb-sec). By applying the maximum torque in a manner to produce forward and reverse saturation of the CMG's against the librating motion, the 36,000-kg example system above could be damped from 0.4 deg/sec to null in 175 orbits or 11.7 days.

Thus, it is concluded that (1) some form of libration damping will likely be required to assure safe revisit conditions can be met, and (2) although additional analyses are required, preliminary looks at the magnitude of the required damping indicate no serious impacts on construction design or logistics.

3.2 THERMAL CONTROL

As a part of the overall support services analysis related to space construction, two thermal control issues were briefly analyzed. The objective is to develop an initial understanding of the relative importance of thermal control to the space construction problem and to determine if there are any critical factors or problems which could affect the space construction process. The two thermal control issues are: (1) construction interference with orbiter heat-rejection capability, and (2) space "cold soak" of sensitive elements installed during the construction process but before they are powered up and activated. The first of these issues is also a Shuttle impact consideration and thus is presented in another volume (*Task 3, Construction System Shuttle Integration*) SSD 79-0124. The "cold soak" analysis is presented in the following paragraphs.

3.2.1 Cold-Soak Analysis

During the construction of large structures in space, it will be necessary to mount certain components in place well in advance of their activation. For those components which have temperature limitations, there is the possibility that survival could be threatened by a long "cold soak" in space. Active spacecraft avoid such dangers by a variety of thermal control techniques. It is the purpose of this study to determine what measures, if any, will be required to protect inactive components during lengthy periods of construction.

The temperature history of the component is a function of many variables, which make a general solution difficult. The approach taken here is to study one component—the RCS module—in some detail, and then generalize the results for other components.

Temperature History of RCS Module

For all large space systems considered in this study, the RCS modules are located at beam extremities on both ends of the structure. This configuration is likely to be assumed for large linear structures because it minimizes problems due to plume impingement on adjacent surfaces and provides maximum control moments for a given fuel expenditure. Accordingly, long-term shadowing of the modules by parts of the structure will not occur. The principal cause of temperature excursions will be the periodic sequence of sunlight and shadow produced by orbital motion around the earth.

Gravity-gradient effects will tend to orient a long structure in an earth-pointing direction. Such an attitude would produce the largest temperature gradients in a component since its "top" or outer surface would be exposed alternately to full sunlight and deep space, whereas the "bottom" surface would see the earth's radiation with or without reflected sunlight (albedo).

It is not at all certain that the structure would maintain an earth-pointing attitude, however, Recent studies using the VARMAP program (which computes the motion of a variable-mass structure in orbit) show that a typical linear structure being constructed from the orbiter can swing and tumble in low earth orbit. Such motions would tend to smooth out temperature excursions. However, an earth-pointing attitude was assumed for the purposes of making a worst-case thermal analysis.

The RCS module and its simplified thermal model are shown in Figure 3.2-1. The largest thermal masses in the system are the oxidizer (nitrogen tetroxide) and fuel (monomethy hydrazine) tanks, weighing 882 kg and 1412 kg, respectively, when full. The thermal capacitance of secondary tankage and structure was neglected. Both tanks are enclosed by a micrometeroid shield which also functions as a radiation shield. In this analysis, the shield was assumed to be so thin that its thermal capacitance could be neglected.

The simplified model assumes that only the upper half of the shield receives appreciable direct solar radiation, and only the lower half receives earth radiation and reflected solar radiation. Both upper and lower shields lose heat to space. The propellant tanks exchange heat with the shields by radiation and with each other by a combination of radiation and conduction. It is assumed that tank and inner shield surfaces are blackened to promote radiation exchange within the compartment. Emissivity of the outer shield surface was taken as 0.4, typical of a rough aluminum solar surface. Solar absorptivities of 0.4 (bright metal) and 0.8 (heavily anodized metal) were assumed as limiting cases.

Figures 3.2-2 and 3.2-3 show initial transients and steady-state temperatures for a 100-minute low-earth orbit. Bright and anodized shield surfaces were evaluated, each starting at a uniform temperature of 300 K. Both solutions have the same characteristics. Tank temperatures go through a low-amplitude oscillation in temperature while slowly tending toward their steady-state values. Actually, the limiting values are not steady, but periodic about orbital average temperatures. These limiting temperatures are found by solving the approximate equations.

$$(1)\quad T_1^{\,4} = K\,(T_2 - T_1)\,\frac{[1 + (FA)_3\,\varepsilon/(FA)_1]}{\sigma\,(FA)_3\varepsilon} + \frac{A_2\,\alpha\,\overline{S}}{\sigma\varepsilon\,(FA)_3}$$

$$(2)\quad T_2^{\,4} = K\,(T_1 - T_2)\,\frac{[1 + (FA)_3\,\varepsilon/(FA)_1]}{\sigma\,(FA)_3\varepsilon} + \frac{A_2\,\varepsilon\,\overline{E}}{\sigma\varepsilon\,(FA)_3}$$

Here, K is the effective conductance between tanks

$(FA)_1$ is the exchange factor between tank and shield

A_2 is the projected area of the shield

$(FA)_3$ is the exchange factor to space

$\overline{S}$ is the orbital average direct solar radiation

$\overline{E}$ is the orbital average earth radiation

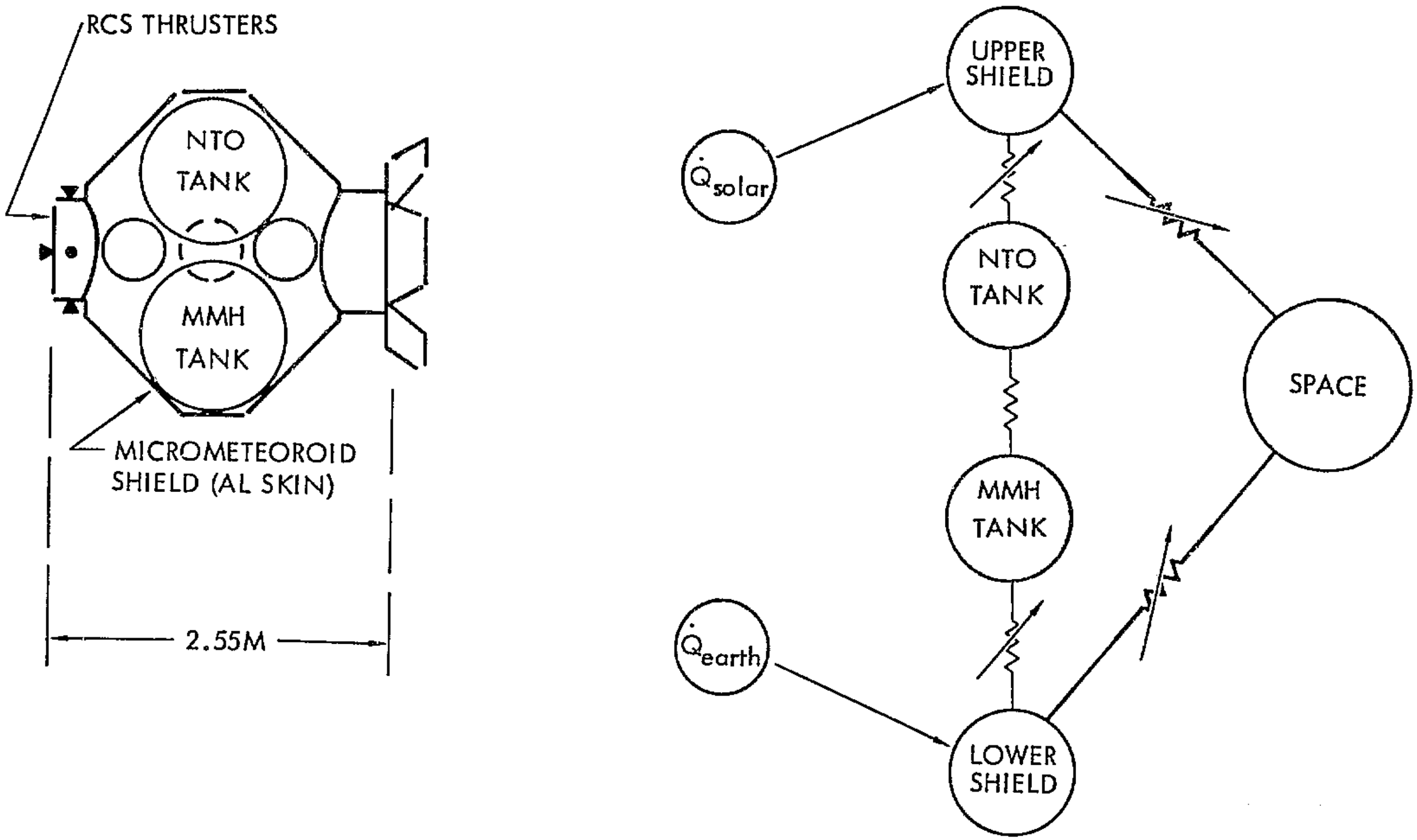


Figure 3.2-1 Simplified Thermal Model for RCS Module

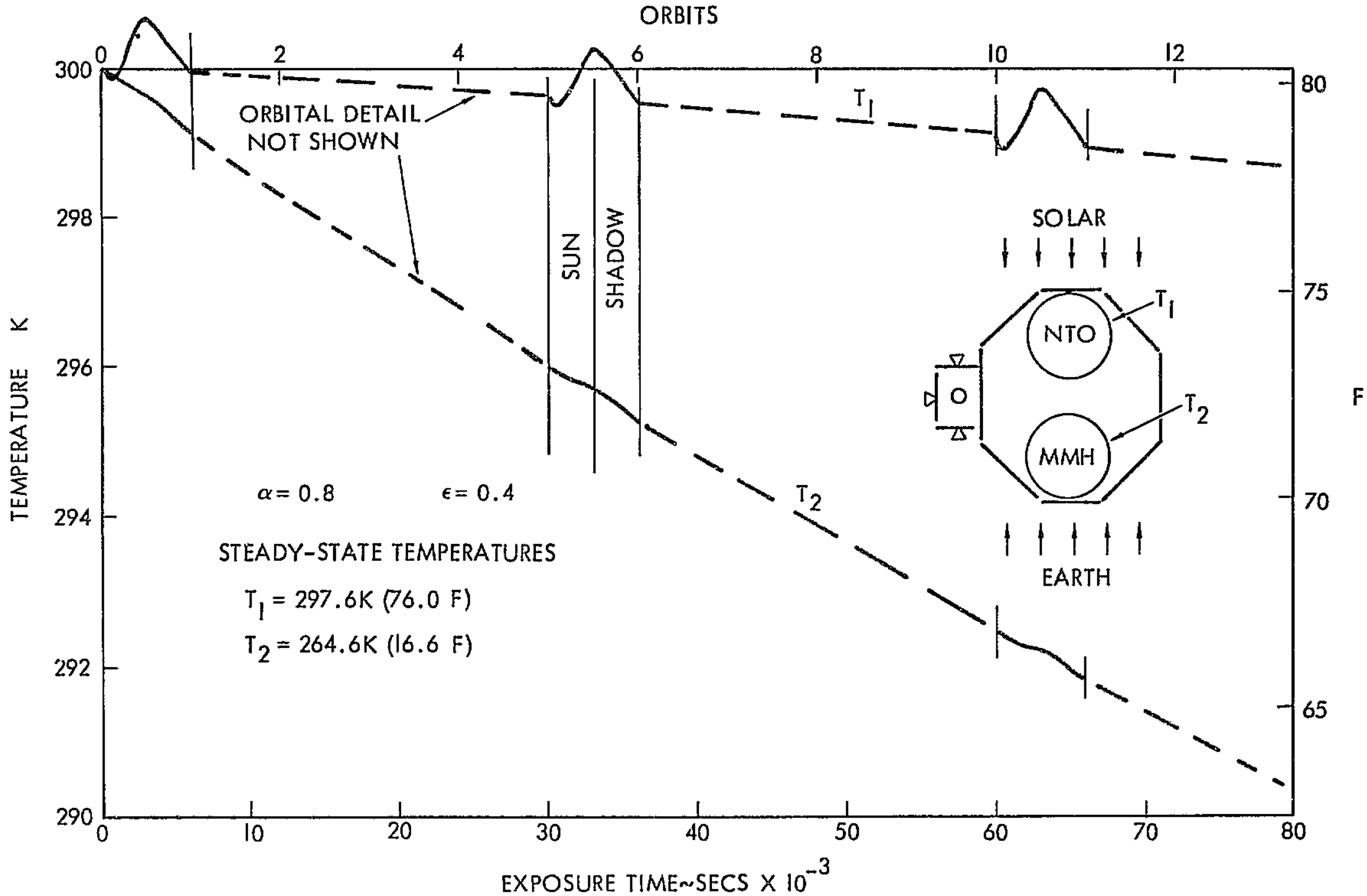


Figure 3.2-2 RCS Module Transient Temperatures (Anodized Skin)

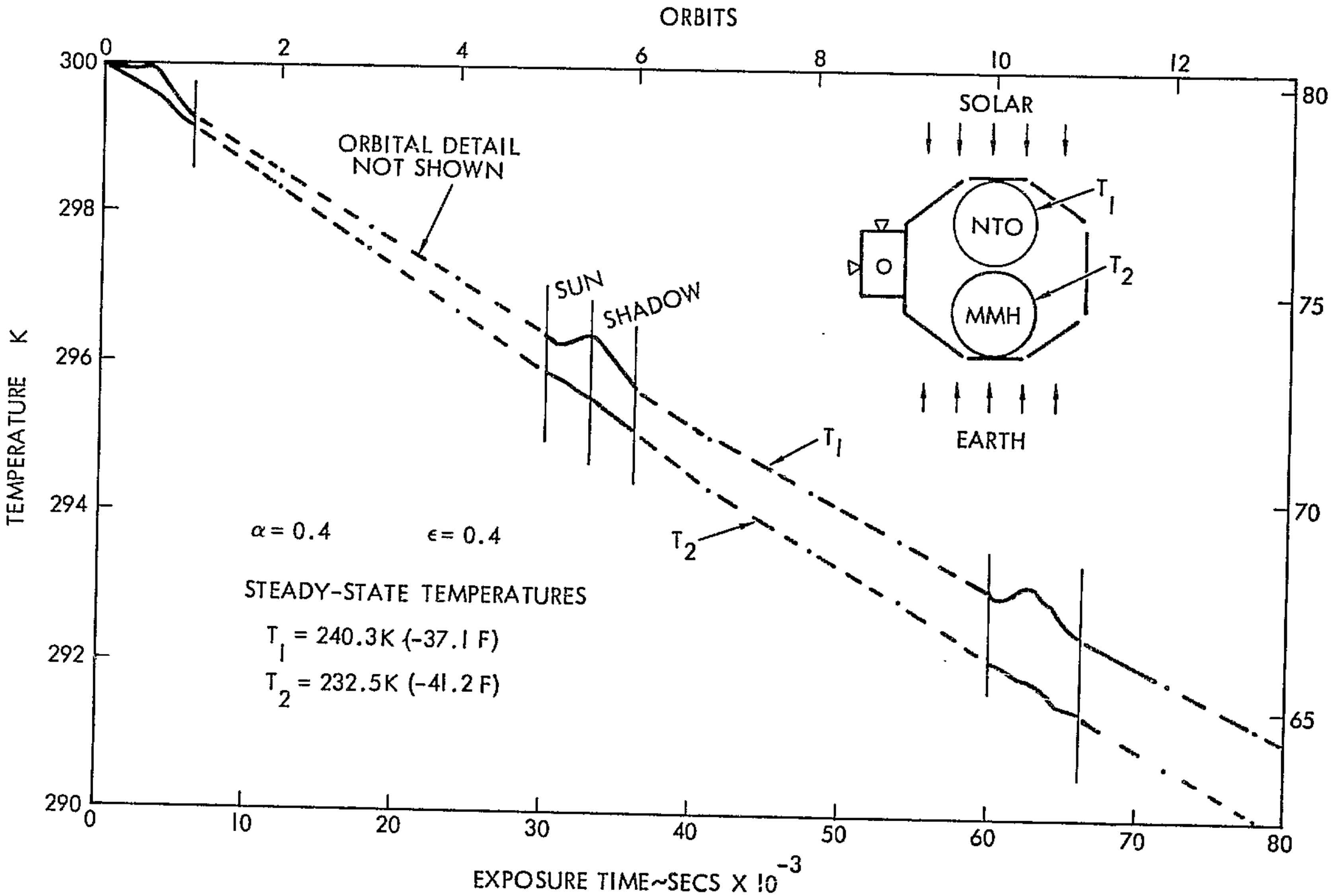


Figure 3.2-3 RCS Module Transient Temperature (Bright Skin)

Table 3.2-1 compares these limiting temperatures with propellant freezing and boiling points (an indicator of survival limits). It appears clear that an appropriate choice of surface properties on the micrometeoroid shield could easily maintain both propellant tanks within their survival limits during long period of exposure in space.

Table 3.2-1. RCS Propellant Survival Temperatures

Propellant	Freezing Point	Boiling Point	Steady-State Temperatures $\alpha = 0.8$	Steady-State Temperatures $\alpha = 0.4$
Nitrogen tetroxide	249 K	294	295 K	240 K
Monomethyl hydrazine	221	361	2[illegible]5	233

Survival of Nickel-Hydrogen Battery Packs

The detailed configuration of battery packs for large space systems has not been defined as well as in the case of the RCS modules. Preliminary thinking visualizes the batteries as box-like modules, each with an integral heat-pipe radiator for heat dissipation. Individual modules would be mounted close together in rectangular racks so that the outline would be a large parallelopiped with protruding fin surfaces. The battery pack would be located at the edge of the space structure, free of significant shadowing by adjacent structure. Like the RCS module, the battery pack as a whole would have a large thermal capacitance. For example, the battery pack for the SPS test article weighs 964 kg.

Based upon the foregoing description and results for the RCS module, the following estimates have been made concerning battery survivability. It seems likely that internal conduction in the pack will be better than within the RCS module. The temperature gradients from "top" to "bottom" of the unit will be less. This fact plus the large thermal capacitance of the component ensures that the battery pack will slowly approach a nearly uniform temperature consistent with orbital average conditions. If it were not for the presence of the radiators, this orbital average temperature would be similar to the average temperature established for the RCS module. With radiators attached, however, there are complications.

From an operational standpoint, the radiator heat pipes on the battery pack should be of the diode or variable-conductance type. However, this could result in dangerously low battery temperatures. The heat pipes would drain heat away whenever the radiators saw a colder environment, but would not replace heat during other parts of an orbit. Ordinary two-way heat pipes would permit both gain and loss of heat. Thus, overall effect would be to increase the effective radiating area of the pack, resulting in somewhat lower average orbital temperature. A more serious consequence would be the possible operational constraints put on the system; for example, orientation, operational windows for charge/discharge, etc.

A better solution appears to be to design the battery radiators so that they could be thermally coupled just prior to operation, rather than integral with the battery structure as now planned. Table 3.2-2 shows estimated battery pack temperatures, based upon the calculated average temperatures for the RCS module with half of the temperature gradient.

Table 3.2-2. Estimated Steady-State Battery Pack Temperatures (Radiator Thermally Decoupled)

Section	Survival Limits Cold[1]	Survival Limits Hot[2]	Steady-State Temperatures $\alpha = 0.8$	Steady-State Temperatures $\alpha = 0.4$
Top	238K	293K	289K	238K
Bottom	238	293	273	234

1. Approximate freezing point of electrolyte
2. Storage limit for long battery life

General Considerations

The preliminary analysis described above suggests that large, exposed, inert components can be maintained in low earth orbit for long periods without exceeding reasonable temperature limits. One-way radiator heat pipes pose a problem, however, when coupled to thermally sensitive components.

Objects small enough to cool significantly in one orbit have not been considered. They must be examined on a case-by-case basis during later design phases.

In general, the components most vulnerable during long period of time in orbit, are those containing fluids which can freeze or build up high vapor pressures. Such components must be systematically identified and given protective measures.

3.2.2 Construction Interference with Orbiter Heat Rejection

See Section 2.7 in the volume of the final report entitled, *Task 3, Construction System Shuttle Integration*, SSD 79-0124.

3.3 ILLUMINATION AND TELEVISION SUPPORT SERVICES ANALYSIS

3.3.1 Objectives and Summary of Study Results

The analytical effort involving illumination services and television has been directed toward the following objectives:

(1) Determine potential electrical power requirements (primarily for darkside illumination) which could drive construction methods, define requirements for illumination equipment and vision aids, and determine construction strategy.

(2) Select general, cost-effective approaches to illumination services, TV equipment, and vision aids for eclipse conditions.

(3) Determine need for attitude control of space construction projects, considering natural illumination and solar power requirements (primarily on sunlit side of orbit), protective shading/diffusing equipment, sensors, and sensitivity of crew and TV equipment.

(4) Develop parametric data and general guidelines for synthesizing illumination/equipment power profiles, crew work timelines, construction strategies and methods, including design of equipment.

(5) Identify potentially effective technology development efforts related to illumination, shading, vision and TV equipment.

As a result of the study effort a general approach to illumination services was developed for space construction projects, an approach which recognizes the complementary but somewhat different needs for protection and vision enhancement for the human eye and for television cameras. Also considered were the integrated systems impacts of natural illumination, thermal control, attitude control, and the lighting-related equipment and procedures for large space construction projects. The major findings and characteristics of the recommended systems approach are summarized in Tables 3.3-1 and 3.3-2. Specific differences in illumination services and television for the three project systems are not readily identifiable nor meaningful, since a large percentage of the recommended approaches are usable for many construction methods and applicable to nearly all critical functions studied. The following discussion material outlines the analytical effort, study results, and rationale supporting summary Tables 3.3-1 and 3.3-2. Also presented are pertinent hardware details and specific design guidelines.

3.3.2 Discussion - General Issues in Illumination and TV Services for Space Construction

Space construction processes are strongly dominated by transport, joining, aligning and inspection functions, which require means to accurately sense orientations and alignments, positions (especially critical clearances), relative velocity and condition of deployment. Experience has shown that the use of direct or aided human vision (telescopes, TV, etc.) to perform a majority of these critical sensing operations is generally cost effective and highly

Table 3.3-1. Summary of Analysis Results and Recommendations for Illumination and TV Services for Space Construction—Dark Side of Orbit

ANALYSIS RESULTS	APPROACH RECOMMENDATIONS

o POWER REQUIREMENTS FOR ILLUMINATION ON ECLIPSE SIDE OF ORBIT CAN BE VERY HIGH
o CONSTRUCTION WORK SHOULD BE CONTINUED DURING ECLIPSE PHASE TO EFFECTIVELY USE CREW TIME
o CURRENT ORBITER LAMPS NOT COMPATIBLE WITH FREQUENT ON-OFF CYCLES.
o NIGHT VISION GOGGLES AND LOW LIGHT LEVEL TV ARE AVAILABLE, NEED FURTHER DEVELOPMENT FOR SPACE CONSTRUCTION
o SOLAR POWER ARRAYS CAUSE UNDESIRABLE HIGH DRAG IN LEO AND REQUIRE SOLAR POINTING (ATTITUDE CONTROL). AFFECTS
    o STRESS LEVELS
    o RCS FUEL
    o CMG ELECTRICAL POWER REQUIREMENTS
	o MINIMIZE ILLUMINATION REQUIREMENTS
	    o CONCENTRATED WORK SPACE, FEW LAMPS
	    o LAMPS ONLY WHERE AND WHEN NEEDED FOR CRITICAL TASKS
	    o PORTABLE, BATTERY POWERED LAMPS FOR REMOTE WORK SITES
	    o RUNNING LIGHTS, FLASHERS AT KEY POINTS
	    o NEW LAMPS: COMPATIBLE WITH FREQUENT ON/OFF CYCLING, SPACE ENVIRONMENT
	    o LIGHT-COLORED, FLAT FINISHES
	    o RETROFLECTORS WHERE APPROPRIATE
	    o REFLECTOR PANELS WHERE FEASIBLE
	    o CHECKOUT EQUIP. IN LIGHTED P.L. BAY
	    o DARK ADAPT CREW AT START OF ECLIPSE PERIOD/REST TIME
	    o LOW-LIGHT-LEVEL TV
	    o NIGHT VISION GOGGLES AND SCOPES-CABIN CREW AND EVA CREW
	    o INDIRECT LIGHTING FOR TV
	o AS LAST RESORT, INSTALL SOLAR POWER ARRAYS EARLY IN CONSTRUCTION SEQUENCE TO PROVIDE MORE POWER FOR ILLUMINATION.

Table 3.3-2. Summary of Analysis Results and Recommendations for Illumination and TV Services for Space Construction—Sun Side of Orbit

ANALYSIS RESULTS	APPROACH RECOMMENDATIONS
o ACTIVE ATTITUDE CONTROL IS UNDESIRABLE AND PROBABLY NOT REQUIRED o AVOIDS RCS IMPULSE LOADS o MINIMIZES FUEL, STABILIZE & CONTROL COMPLEXITY o UNDESIRABLE VIEWING CONDITIONS SHORT-LIVED	o UTILIZE GRAVITY GRADIENT, AERO. DRAG ORIENTATION FOR FAVORABLE RANGE OF ORIENTATIONS. o FAVOR USE OF EARTH SHINE ILLUMINATION; ALSO FAVORABLE FOR BACKGROUND VIEWING
o TV CAN ACCOMMODATE TO EXPECTED BRIGHTNESS RATIOS EXCEPT FOR DIRECT VIEW OF SUN OR SOLAR REFLECTIONS ON SPECULAR SURFACE	o PROVIDE FILTERS FOR EXCESSIVELY BRIGHT CLOUDS, SURFACES o PROVIDE FLAT FINISHES, COLOR CONTRAST o DESIGN CONSTRUCTION PROJECT, FIXTURES AND EQUIPMENT TO REDUCE DIRECT VIEW OF SUN OR ITS REFLECTION IN SPECULAR SURFACES.
o DIFFUSE LIGHT REFLECTED FROM EARTH, ORBITER BODY AND WING IS HELPFUL.	o PROVIDE AUTOMATIC IRIS CUTOFF ON TV CAMERAS, ALTERNATE TV CAMERAS
o ACTIVE CONTROL, TILT AND PAN DESIRABLE FOR RMS END EFFECTOR TV AND LIGHTS, REMOTE TV'S ON STRUCTURE, OR CONSTRUCTION FIXTURE	o PROVIDE TILT AND PAN, REDUNDANT TV o PROVIDE LOCAL SHADES, DIFFUSERS FOR EVA CREW, TV CAMERAS

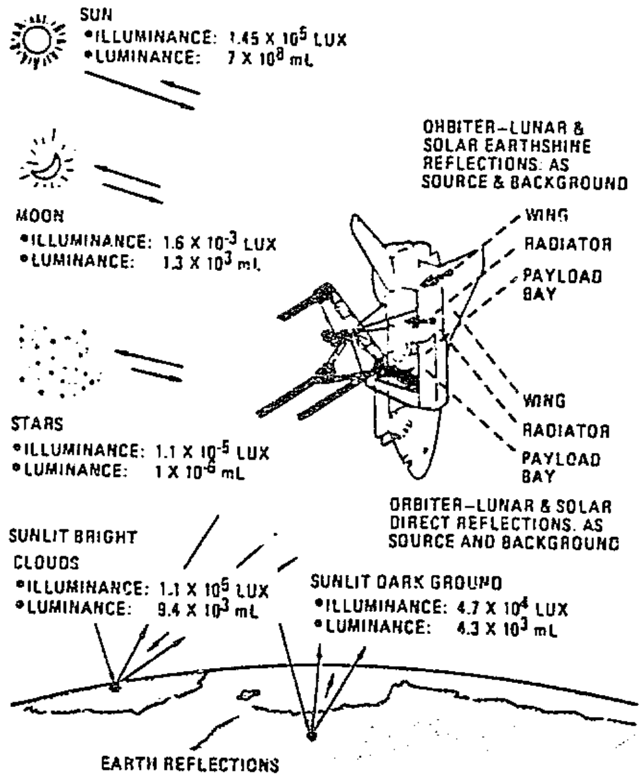
reliable when adequate illumination is provided. However, preliminary studies have identified several potentially significant problems associated with assuring adequate illumination in space, whether the light is artificially provided or naturally received. In particular, low-earth orbits (LEO) involve consideration of frequent changes from a very dark environment to a brightly sunlit environment (but with deep shadows in many cases), which in turn create a concern about work interruption due to necessities of adaptation by both man and machine (TV camera). Figure 3.3-1 provides pertinent data on the wide range of illumination fluxes and brightness conditions in LEO.

The low earth orbit situation envisioned and the peculiar stark contrasts of deep shadow and bright sunlight in space require special lighting provisions for vision to be adequate. However, there are differences between the various methods which affect the required intensity for lighting, the location of the lamps and the resulting power, weight, volume and cost of the lighting and vision aspects.

Table 3.3-3 presents an attempt to outline the most important considerations of the subject. With this context in mind, the following discussion separately considers the dark (eclipse) side of low earth orbit, in which artificial lighting considerations dominate, and the sunlit side of orbit, primarily involving direct solar illumination and multiple reflections.

3.3.3 Power Requirements for Artificial Illumination and TV on Dark Side of Orbit

Previous studies of space construction have determined that the power requirements for lighting on the dark side of LEO may constitute a significant portion of the total energy requirement for space construction. For example, a study by Grumman (Reference 1) indicated that lighting could require very large amounts of power, from 35 kW to 83 kW, depending on the size of the project. Further data on these studies of power requirements for differing situations appear in Figure 3.3-2. Actually, such power demands are well within the capacity of power output from solar arrays for each of the three construction projects studied during this Rockwell Systems Analysis study (Reference 2). However, in order to obtain such power, the large solar arrays and storage batteries would have to be carried up, deployed, and connected to the construction fixture lighting system and other necessary lamps early in the construction sequence. Such solar arrays would need to be oriented so as to face the sun during the sunlit period. Large areas of low mass, such as solar arrays, can cause significant drag on the orbiting construction project at low earth orbit altitudes. This combination of effects is generally undesirable as regards fuel usage for stationkeeping and for pointing on orbit. Also, loads on weakly supported structure during the construction processes can be a significant problem. The SPS Test Article project represents a case of project design where the solar array blankets cannot be deployed until the structure is at least partially built. In such a situation, the lighting power must be limited to that available from the orbiter or from other auxiliary power sources until some of the space construction project blankets are installed and connected to a service power system. As a result of such considerations, it is recommended that lighting systems for space construction should be frugal in power demand.



Characteristics Affecting Space Construction

- Wide Range of Brightness - Stars to Full Sun: 10^{10}:1
- Lighting Conditions Continually Changing Angle, Light, Dark
- Eye Does Not Fully Dark Adapt During Eclipse Phase of LEO
- Shadows Generally Darker Than in Earth Atmosphere, Structure Outlines May Be Obscured
- Orbiter is Helpful as a Diffuse Light Reflector
- Bright Clouds Reflection and Direct Sun Are Similar in Illumination Intensity But Clouds More Diffuse
- Solar Orientation is Critical - Avoid Looking at it

Figure 3.3-1 Natural Illumination Environment-LEO

Table 3.3-3. Summary of Viewing and Illumination Considerations for Space Construction

- Relatively rapid cycling from bright sunlight to darkness at low-earth orbit.
  - Power and lamp cycling questions (life, surge, thermal)
  - Dark adaptation considerations of crew eye protection
  - TV range of sensitivity (brightness ratios)
- Power requirements for darkside illumination
  - Overall lighting versus local lighting
  - Reflective, light colored surfaces
  - Portable versus fixed lamps
  - Continuous lighting versus cycled lighting
- Optimum viewing angles for sunlit side viewing
  - Glare avoidance - direct, veiling, contrast
  - Backlighting problems of earth and sun
  - Diffusivity of surfaces versus glare
  - Vision requirements versus thermal requirements
  - Orientation of spacecraft and crew for self-protection from disturbing light input
  - Possible use of shades and diffusers
- Vision angles required by configuration
  - View from crew compartment windows
  - TV camera viewing angles and positions
  - EVA viewing positions and angles
- Work interruptions versus productivity
  - Dark versus day
  - Viewing cutoff by glare, intensity of light, angle, shadows
- Interaction with thermal consideration
- Interaction with communications considerations
- Interaction with stability/control considerations
- Interaction with orbital plane requirements, launch time, season and relative rate of change of β angle
- Hardware selection and power requirements
  - Incandescent lamps
  - Metal Halide
  - Beam versus Flood
  - Reflectors
  - Finishes
  - Portable versus fixed
  - Solar power versus batteries
  - Orbiter fuel cells versus outside sources
- Assembly sequence versus power requirements
  - Setup power module/solar array before construction
  - Setup solar array at end of construction
  - Schedule operations requiring significant illumination to be performed on sunlit side of orbit.

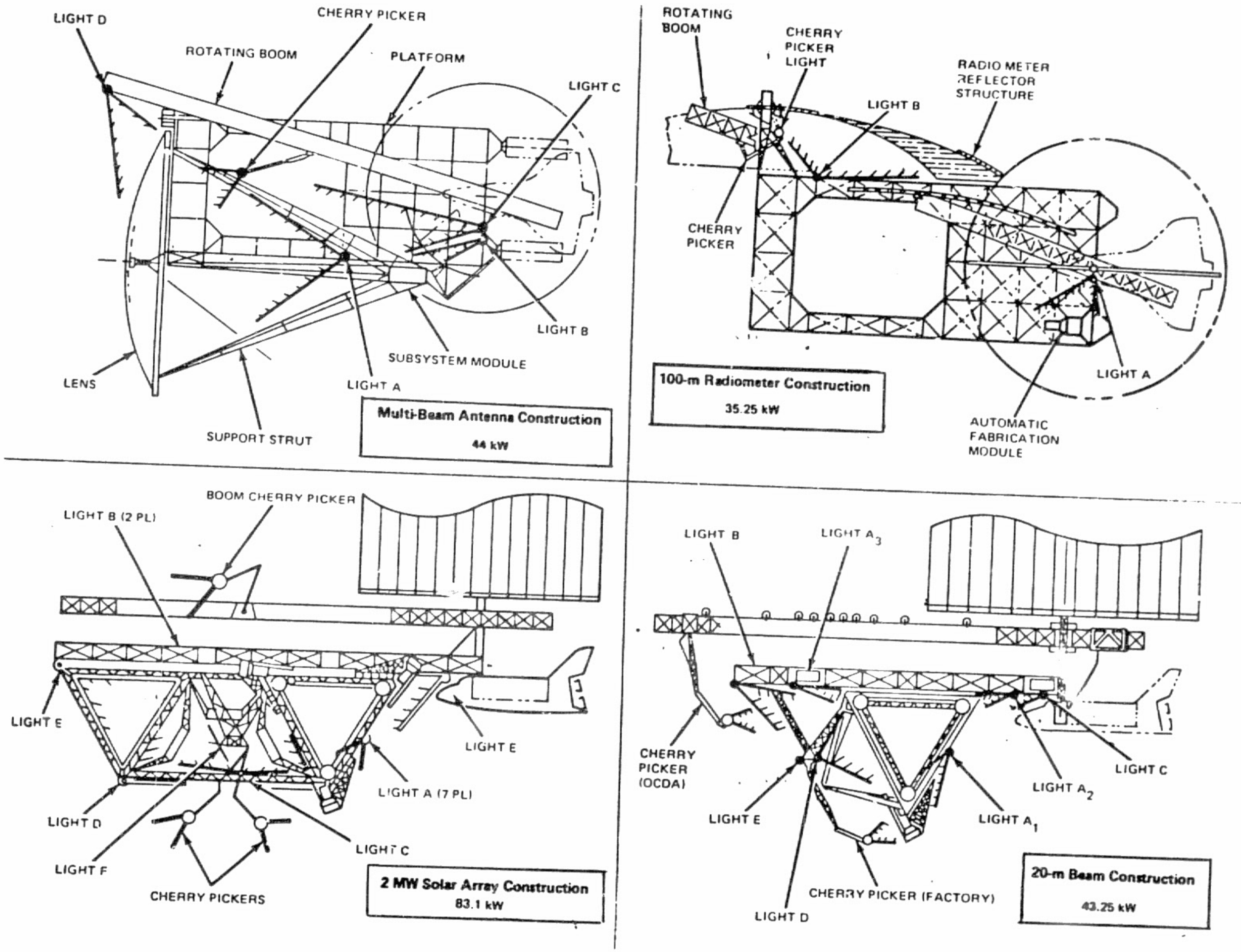


Figure 3.3-2 Large Space Construction Project Illumination Power Requirements - OCDA Approach

Satellite Systems Division
Space Systems Group
Rockwell International

A question thus arises, "Is it feasible to significantly reduce power demands for illumination?" Examination of the basis for the Grumman study indicates that it is, because the study was based on a relatively generous, general and continuous flood-lighting approach, such as one might use for factory or shipyard construction site illumination. In contrast, a study by Rockwell International took a more austere and limited approach. The concept involved the use of carefully controlled, localized lighting for fine, close work and the use of reflectors, small "running lights", and various portable lamp devices as methods to control and minimize lighting power. The study involved smaller construction projects (two versions of an electronic mail satellite) and a larger number of options in differing types of construction facility and methods (manual versus remote, automatic). Selected cases, illustrating the general variety of satellite shapes, construction fixtures and facilities, are shown in Figure 3.3-3. The continuous power requirements for illumination used in these various cases are summarized in Table 3.3-4. To set these data in perspective, power and energy for lighting is compared to other significant power and energy demands for several cases in Tables 3.3-5, 3.3-6 and 3.3-7. Note the relatively large percentages of energy required for illumination. Even these austere lighting concepts required power levels from 2.0 kW to 4.8 kW, which represents from 38 to 52 percent of the total system construction power requirements. Such levels of power, when combined with power from normal operations, RMS, etc., seriously tax the Shuttle Orbiter capability and indicate a potential advantage in searching for even more efficient low-power lighting concepts.

Concepts for truly minimal illumination power do exist. A rather extreme, but promising approach is to use electronic amplification of available illumination (natural and artificial), such as employed by low-light level TV cameras and by military night vision goggles and scopes (Figures 3.3-4 and 3.3-5). Carried to their logical extremes, such systems could drastically reduce, perhaps even eliminate the need for artificial lighting altogether. The naturally available starlight and moonlight might then be sufficient! Clearly, some additional development would be required to use such goggles and scopes in space applications. Yet the approach seems promising, since the major electronics feasibility has been proven (References 3 and 4).

The potential benefits would not be without some detriments such as the following:

(1) Development Costs - Space rating of night vision goggles for use in the crew cabin and stowage provisions required; blackout curtains around windows; re-configuration of a night vision system for EVA usage, including helmet integration (interior or exterior?), power supply, donning/doffing and stowage provisions.

(2) Lack of Color Vision - Monochrome, green image.

(3) Color Vision "Fatigue" - Crew eyes may see brown after-images (from green phosphor color in night vision scope).

(4) Somewhat Reduced Depth Preception - Especially with single scope, but partially due to optics involved.

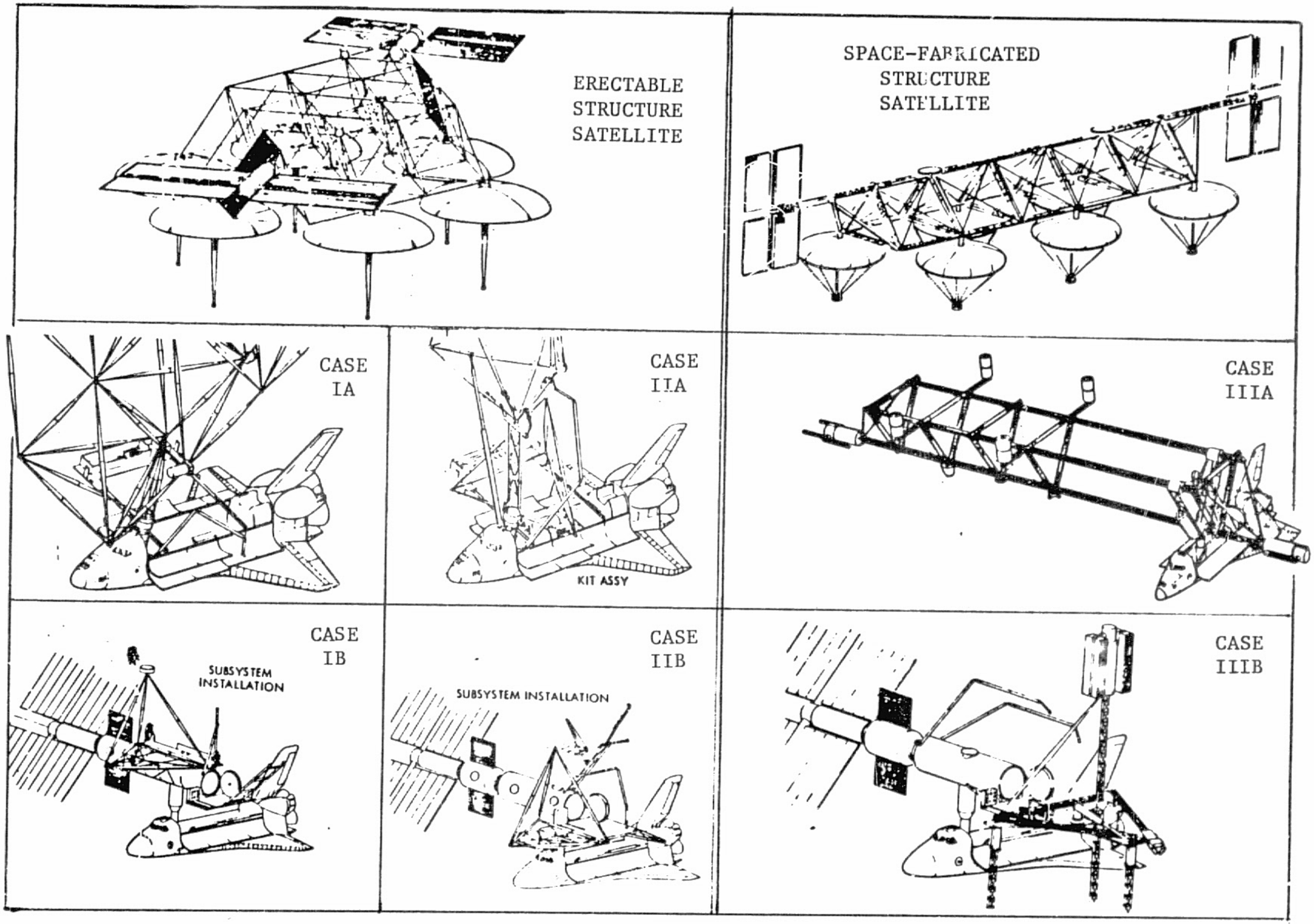


FIGURE 3.3-3 Electronic Mail Satellite Construction Projects and Approaches

Table 3.3-4 Illumination Power Requirements for Several Space Construction Approaches - Electronic Mail Satellites

FUNCTION/LOCATION	POWER REQUIRED (WATTS)							
	I		II		III		IV	
	A	B	A	B	A	B	A	B
BASIC ORBITER	1400	1400	1400	1400	1400	1200		1200
RMS	200	200	400	200	400	200		200
CONST. FACILITY OR PAYLOAD	960	1240	960	1560	1800	1400		2840
TOTAL (ALL OPERATING)	2560	2840	2760	3160	3600	4800		4240

Table 3.3-5. Energy Requirements for Electronic Mail Satellite (EMS) Space Construction, Case IA - EVA

Item	Power (kW)	Flight 1 Energy (kWh)	Flight 2 Energy (kWh)	Flight 3 Energy (kWh)
(1) Construction lighting	2.56	239	239	264
(2) Charge EMU suit (2)	2	150	150	156
(3) Charge MMU (2)	0.4	30	30	32
(4) Heaters - RMS	0.8	16	16	21
(5) RMS	1	15	15	28
(6) Construction equipment checkout	1.1	3	3	3
(7) System checkout	1.5	9	9	45
Total required	-	462	462	549
Lighting % of total	-	52	52	48
Energy available + cryo kit	-	50 840	50 840	350 658 (840)
Total available	-	890	890	1008

Table 3.3-6 Energy Requirements for EMS Space Construction, Case IIA - Manipulators

Item	Power (kW)	Flight 1 Energy (kWh)	Flight 2 Energy (kWh)	Flight 3 Energy (kWh)
1) Construction lighting	2.76	285	285	219
2) Heater - RMS	0.8	82	82	64
3) RMS	1.0	93	93	78
4) Heater - construction manipulator	1.0	102	102	-
5) Construction manipulator	1.2	80	80	-
6) Checkout - construction equipment	1.1	6	6	-
7) Checkout - system	1.5	8	8	63
Total required	-	656	656	424
Lighting % of total	-	43	43	52
Energy available + 1 cryo kit	- -	650 190 (840)	650 190 (840)	650 -
Total available	-	840	840	650

Table 3.3-7 Energy Requirements for EMS Space Construction, Case IIIA - One Beam Machine

Item	Power (kW)	Flight 1 Energy (kWh)	Flight 2 Energy (kWh)	Flight 3 Energy (kWh)
1) Construction lighting	3.6	36	231	318
2) Heaters (2) for manipulators	1.8	9	144	227
3) Manipulators (2)	2.2	11	141	167
4) Construction checkout	1.1	9	-	-
5) Beam machine checkout	6.0	3	24	8
6) System test	1.5	-	62	72
Total required	-	68	602	792
Lighting % of total	-	53	38	40
Energy available + 1 cryo kit	- -	168 -	650 -	50 840
Total available	-	168	650	840

2nd Generation Image Intensifier AN/PVS-5

TYPICAL IMAGERY

TARGET RANGE: 50 M
ILLUMINATION: 1/4 MOONLIGHT

SYSTEM CHARACTERISTICS

WEIGHT: 1.9 LBS
MAGNIFICATION: 1 x
STARLIGHT RECOGNITION RANGE: 50 M (MAN TARGET)
FIELD OF VIEW: 40^0

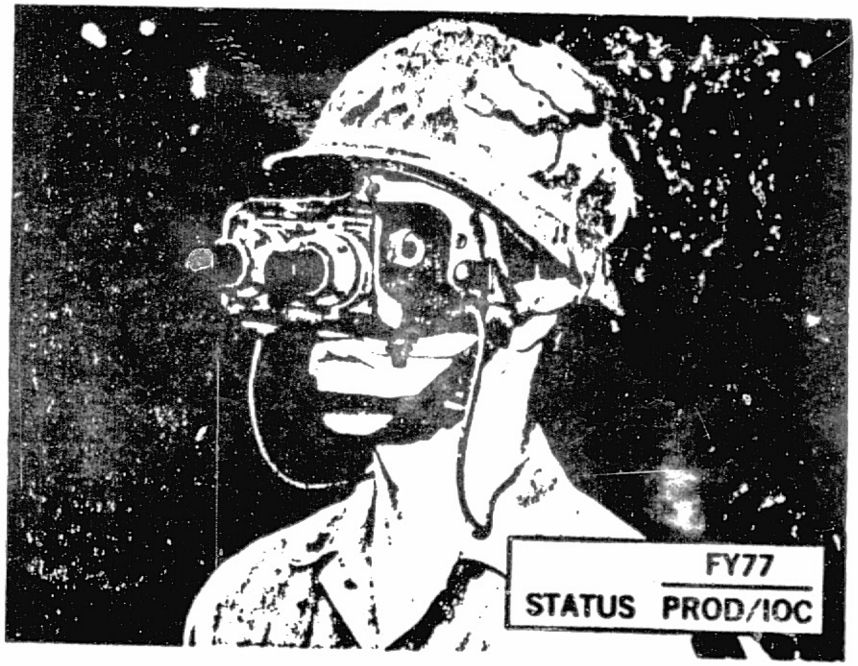
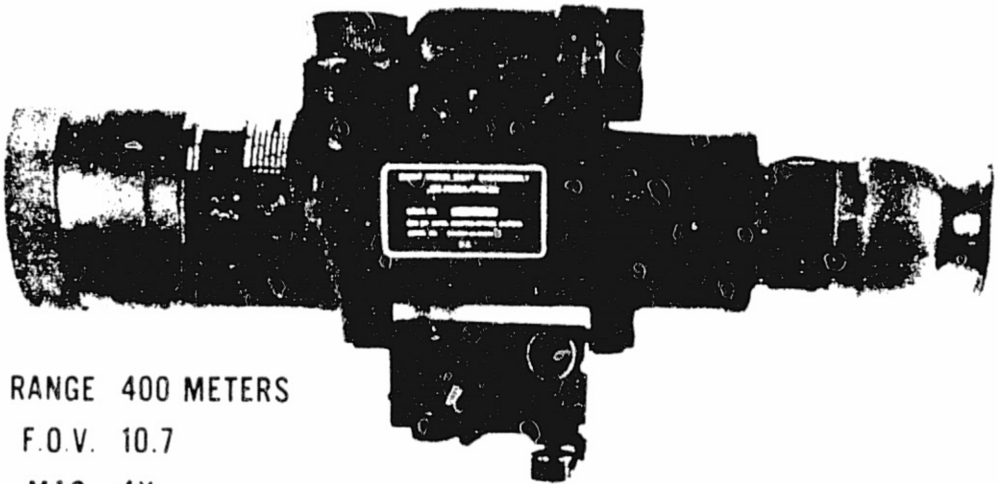


Figure 3.3-4 Night Vision Goggles for Ground Operations

FIRST GENERATION SMALL STARLIGHT SCOPE
AN/PVS-2



RANGE 400 METERS
F.O.V. 10.7
MAG. 4X
WT. 5.75 POUNDS

SECOND GENERATION SMALL STARLIGHT SCOPE
AN/PVS-4

RANGE 400 METERS
F.O.V. 15
MAG. 3.8X
WT. 3.5 POUNDS

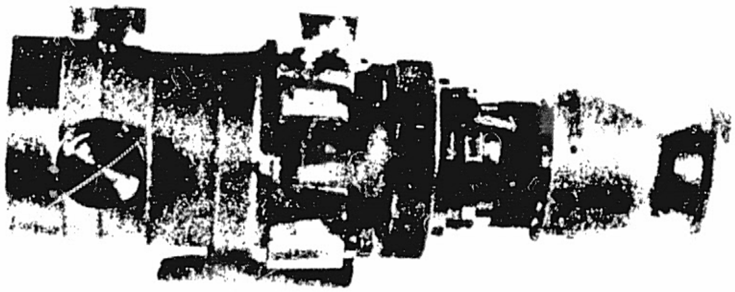


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Figure 3.3-5 Night Vision Scopes for Ground Operations

C - 2

(5) Potentially Limited Acuity - Due to nature of photomultiplier tube resolution.

(6) Partial Dark Adaptation - Desirable for persons using night vision goggles; requires some pre-usage period with minimal lighting and, therefore, represents work interruption time.

The use of light amplification devices during the eclipse phase of orbit brings to the fore a secondary concern, that of orientation considerations for space construction. (In sunlight, orientation is a primary concern, to be discussed later). The naturally available starlight is comfortably diffuse and wide spread, covering over half of the spherical angles surrounding the orbiter/construction project. Moonlight, however, is highly directional and relatively bright compared to starlight, especially at full moon. Shadows in moonlight could become a significant consideration. However, the use of artificial lighting at relatively low levels could practically eliminate any orientation problems for dark side operations. This is in contrast with the sunlit operations, where competing with sunlight is impractical for artificial lighting.

Work Cessation on Dark Side of Orbit

In view of the foregoing complexities, it is appropriate to raise the question, "Why do any construction work on the eclipse side?" If no work is done, no lighting power is required. Closely related to this question is scheduling of unproductive (but necessary) rest periods which can be scheduled for this time, thus mitigating the effects of work cessation or at least minimizing lighting power duration.

Figure 3.3-6 depicts eclipse durations at 463 km altitude versus β angle (angle between orbit plane and sunline). Eclipse durations are about 28 to 36 minutes at 463 km (250 nmi) and a 28.5 degree (52 degree max β angle) inclined orbit. Thus, a ten-minute rest period represents about 36 to 28 percent of the period of darkness. However, the remaining amount (18 to 26 minutes) represents about 19 to 28 percent of the total orbit period. Such percentages are not negligible. EVA operations are especially important as regards maximum use of the available work time, since only about 5 hours out of a 10-hour work day are really useful. The suit donning and doffing time absorb the balance. For efficient use of crew members in the cabin or in EVA, rest periods should be no longer than actually necessary in order to get the maximum benefit of the highly limited work time. On the other hand, a 10-minute rest period could be creatively used for dark adaptation of crew member's eyes in preparation for more efficient vision. Such an adaptation period is generally compatible with effective use of night vision goggles. Alternately, better vision could be obtained for effective use of minimal lighting with the unaided eye.

The foregoing considerations suggest the following general guidelines:

(1) Construction work should not cease during the entire eclipse phase of an orbital period due to lack of adequate illumination unless a very short work period is required and/or other schedule factors or limited power conditions are predominant.

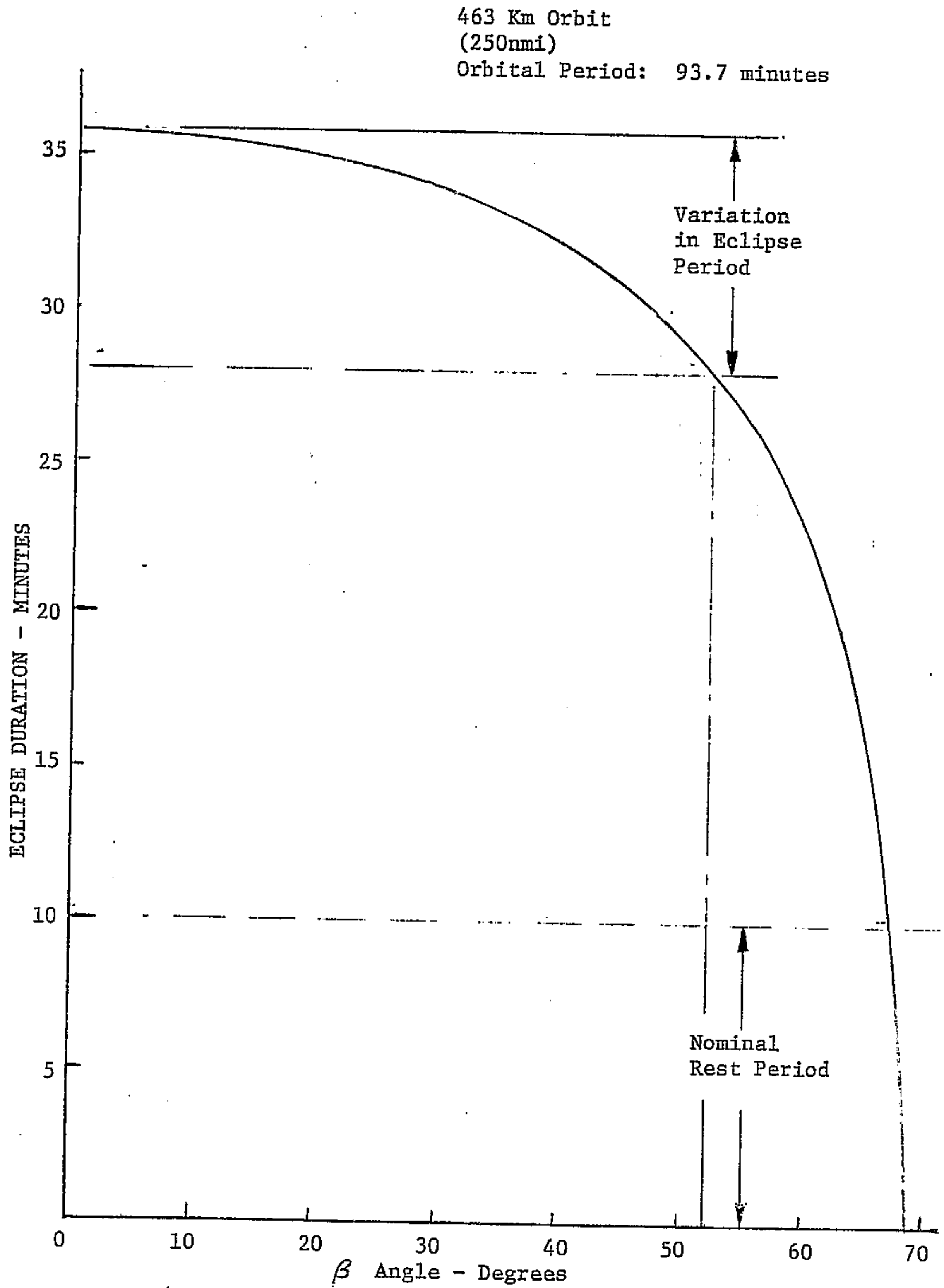


Figure 3.3-6 Duration of Eclipse Period Vs. β-Angle

(2) Power requirements for lighting should be minimized by all reasonable means, particularly for the early phases of construction, where additional power cannot be brought on line until construction is well along.

Specific Light Level Requirements and Light Sources

Having established the above concepts, further detailed evaluation of "reasonable means" for minimizing power requirements for lighting is appropriate. Potential candidates are listed in Table 3.3-8. To definitize these general concepts and approaches, it is now appropriate to examine some specific detail hardware designs and numerical requirements to be met during implementation. Table 3.3-9 lists some typical minimum illumination level or brightness requirements for interior and EVA operations and for television viewing, based on normal vision. It is presumed that NASA recommendations for task lighting account for the helmet and window glass transmittance of approximately 80 percent, but use of any recommendations based on earth situation experience with the naked eye should consider the glass transmittance limitations of the orbiter or other manned work station transparencies (additional detail on transmittances is presented later). Table 3.3-10 provides some typical lighting level recommendations for large, earth-based, open-space facilities, which can be considered analogous to space construction. Obviously, these levels are only applicable for direct vision, and could be reduced if some type of vision amplification is available.

Space construction will require several types of lamp options for specific and general applications. Some of these requirements may be satisfied by use of existing lighting in the orbiter, others by adaptations of lamps developed for the orbiter, Skylab, or even Apollo. Table 3.3-11 suggests several such lamp options for space construction. Figure 3.3-7 depicts the general locations of the six lamps in the Shuttle Orbiter payload bay sidewalls, as well as the two lamps located on the forward bulkhead and top of the crew cabin. Additional location coordinates detail is presented in Figures 3.3-8 and 3.3-9. Note that the six lamps mounted on the payload bay sidewall are very likely to be partially obscured by items stowed in the bay, especially at the beginning of the unloading period in space.

Figure 3.3-10 illustrates two kinds of illumination proposed as standard provisions for the Manned Maneuvering Unit (MMU). These are potentially available for EVA operations in space construction. The major items of concern are the work lights located over the shoulders of the astronaut. With both lights on, 25 watts are expended to illuminate the work space immediately in front of the operator. It is expected the resulting light levels should be adequate for most normal eye vision for close-in, manual EVA assembly tasks in space construction (within arms reach of the astronaut). However, this must be verified by observation in mockups for any specific task. Note that the power demand of these lights as regards its impact on power demand profiles for the orbiter (or auxiliary power supplies for construction), is delayed until the period of battery recharge. Also, note that the demand is controlled by the battery recharge rate. Table 3.3-12 provides significant power demand parametrics for the MMU. Again, use of night vision goggles might justify installation of smaller lamps, using less power. On the other hand, providing a stabiliza-

Table 3.3-8. Approaches to Minimize Illumination Power Demands for Large Space Construction

EQUIPMENT	HARDWARE	OPERATIONS
o LIGHT COLORED, FLAT FINISHES, REFLECTIVE SURFACES AT KEY POINTS	o LIGHT COLORED, FLAT FINISHES, REFLECTIVE SURFACES AT KEY POINTS	o SET UP REFLECTIVE SURFACES WHERE FEASIBLE
o PORTABLE, BATTERY POWERED LAMPS FOR REMOTE SITE INSTALLATIONS	o PROVIDE ATTACH POINTS FOR PORTABLE LAMPS	o CARRY AND SET UP PORTABLE LAMPS DURING ECLIPSE PHASE AT REMOTE SITES, LOCAL WORK AREAS.
o SMALL NUMBER OF FLOOD LAMPS AT CONCENTRATED WORK SITE. CONSTRUCTION FIXTURE WITH CLOSE-IN EQUIPMENT LOCATIONS	o DESIGN FOR JOINING, ASSEMBLY, INSTALLATIONS IN SMALL VOLUME WHICH PROJECT PASSES THROUGH	o PERFORM WORK IN SMALL WORK SPACE TO MINIMIZE TRANSPORT & NUMBER OF LAMPS REQUIRED
o PROVIDE RMS AND REMOTELY CONTROLLED SITES WITH LOW-LIGHT-LEVEL TV CAMERAS	o PROVIDE FOR TV CAMERA INSTALLATIONS, LAMPS, POWER AS REQUIRED	o PERFORM REMOTE OPERATIONS USING LOW-LIGHT-LEVEL TV DURING ECLIPSE PHASE OF ORBIT. AVOID EVA IF CHOICE EXISTS.
o PROVIDE FOR CHECKOUT IN LIGHTED PAYLOAD BAY	o DESIGN FOR CHECKOUT IN NORMALLY LIGHTED AREA OF SMALL VOLUME	o SCHEDULE REST PERIODS AND DARK ADAPTATION OF CREW FOR BEGINNING OF ECLIPSE PERIODS, DO VISUAL CHECK-OUT IN PAYLOAD BAY LATER.
o DEVELOP LAMPS & SWITCH SYSTEMS COMPATIBLE WITH FREQUENT ON-OFF CYCLING IN SPACE	o N/A	o TURN LAMPS OFF ON SUNLIT SIDE
o PROVIDE NIGHT VISION SCOPES AND GOGGLES FOR CABIN CREW. PROVIDE BLACKOUT CURTAINS	o N/A	o USE NIGHT VISION GOGGLES AS NECESSARY DURING ECLIPSE PERIOD, IN CONJUNCTION WITH CABIN BLACKOUT AND DARK ADAPTATION/REST PERIODS
o DEVELOP NIGHT VISION GOGGLES FOR EVA CREW		

Table 3.3-9 Space Construction Illumination Requirements

ACTIVITY	LUMINANCE (FT. LAMBERTS)	ILLUMINATION FT/CNDL	ILLUMINATION LUMEN/M^2	REF.
EVA-TRANSFER ROUTE	≥ 1 *	≥ 2	22 **	*SC-L-0002 (REF 11) **MSFC-STD-512 (REF 12)
EVA - WORK STA.	≥ 5 *	≥ 5 **	54 **	
[INTERIOR - GROSS TASKS - INITIAL ENTRY, EGRESS, TRANSLATION]		1 - 3	10.8 - 32**	
[INTERIOR - CASUAL TASKS - ACTIVATING EQUIPMENT, GROSS READING]	2 - 4 (TBD)	3 - 5	32 - 54**	
RMS TV VIEWING		0.1	1.1	
OVERHEAD WINDOW - DOCKING APPROACH	2 - 4 (TBD)	3 AT 30 FT.	32 - 54	MCR 4481 LAMP REQUIREMENTS
DETAIL ASSEMBLY & INDEXING (PREFERRED)	10 - 24	20 - 30	215 - 323	IES ILLUM. STANDARDS,

Table 3.3-10. Typical Exterior Illumination Recommendations for Earth Facilities Analogous to Space Construction (Reference 5)

Area	Minimum Illumination on Tasks (fc)	Minimum Illumination on Tasks (LUX)
Building (construction)		
General construction	10	108
Excavation work	2	22
Building exteriors		
Entrances		
Active (pedestrian and/or conveyance	5	54
Inactive (normally locked, infrequently used)	1	11
Vital locations or structures	5	54
Building surrounds	1	11
Central station		
Catwalks	2	22
Conveyors	2	22
Entrances		
Generating or service building		
Main	10	108
Secondary	2	22
Gate house		
Pedestrian entrance	10	108
Conveyor entrance	5	54
Fence	0.2	2.2
Fuel-oil delivery headers	5	54
Oil storage tanks	1	
Open yard	0.2	2.2
Platforms-boiler, turbine deck	5	54
Loading and unloading platforms	20	215
Freight car interiors	10	108
Lumber yards	1	11
Parking areas		
Self-parking area	1	11
Attendant-parking area	2	22
Piers		
Freight	20	215
Passenger	20	215
Acting shipping area surrounds	5	54
Railroad yards	1-20	11-215
Ship yards		
General	5	54
Ways	10	108
Fabrication areas	30	323

Table 3.3-11 Lamp Options for Space Construction

APPLICATION	DESCRIPTION	DEVELOPMENT STATUS	COMMENT
Orbiter Payload Bay Sidewalls	6 - in sidewalls 200 watts each metal halide	In work for Shuttle Orbiter	Standard item; may require shrouds to avoid direct view by T.V. cameras.
facing aft, on 576 bulkhead	1 - 200 watts metal halide	"	
facing upward, between overhead windows	1 - 200 watts incandescent	"	Also to be used for RMS
RMS, at end effector	1 - 200 watts incandescent	"	See Above
EVA pathways and work stations	See MSFC-STD-512A Guarded incandescent	Skylab usage	Applicable for EVA on construction fixture
MMU, translation and work station with MMU access	2 each MMU, 12.5W each over shoulders of suited astronaut	In work for MMU, but "on hold" until suit lamp selected	Use for EVA with MMU - may revise to "headlight" concept
Construction Fixture assembly stations	Flood type 100 & 200 watts Tilt and Pan controlled remotely.	100 watt type new 200 watt type from Orbiter basic lamp	Adaptation of existing technology planned for 200 W lamps.
Orientation (Running Lights) at extreme boundaries of structures.	o 3.5W, 2.9V, 1.2A o 25W, 28V, .9A o 50W, Flashing 60ppm at 1 millisec dur.	Available from ILC Technology, Inc. Sunnyvale, CA	Adaptations of existing technology.
Portable Worklight handwired	~80^{o} Cone, 100W Incandescent	New Standard Item Needed	For semi-permanent, handwired convenience lighting General Purpose
Portable Worklight Battery Powered	~80^{o} Cone, 10W Incandescent	New Standard Item Needed	For remote site installation & joining
Penlight, hand carried by EVA Astronaut	Small 3 Volt flashlight	Apollo & Skylab Standard Item	For supplementary, close-in, difficult-access worklight requirements

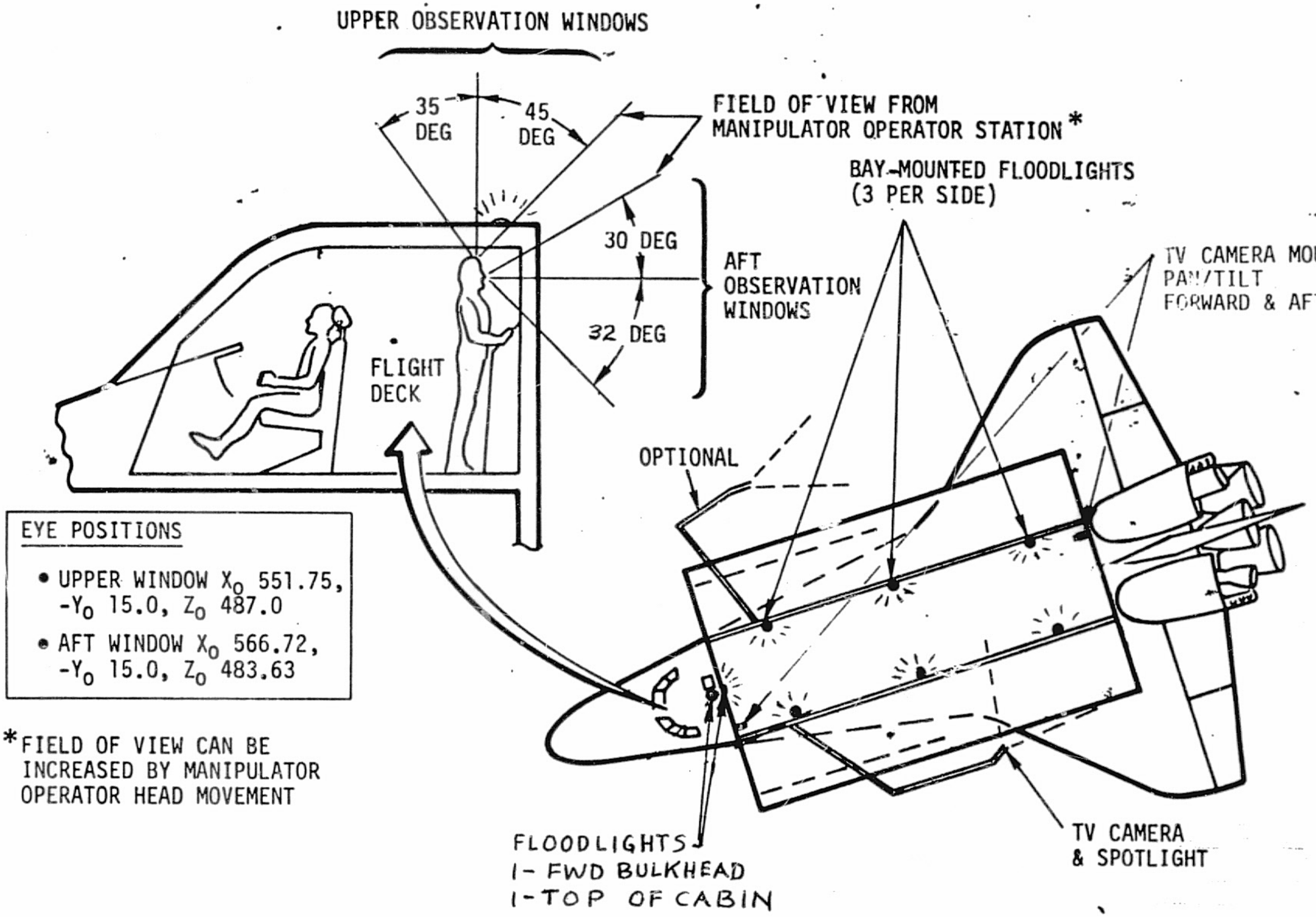


Figure 3.3-7 General Arrangement, Orbiter Lamp and TV Camera Locations

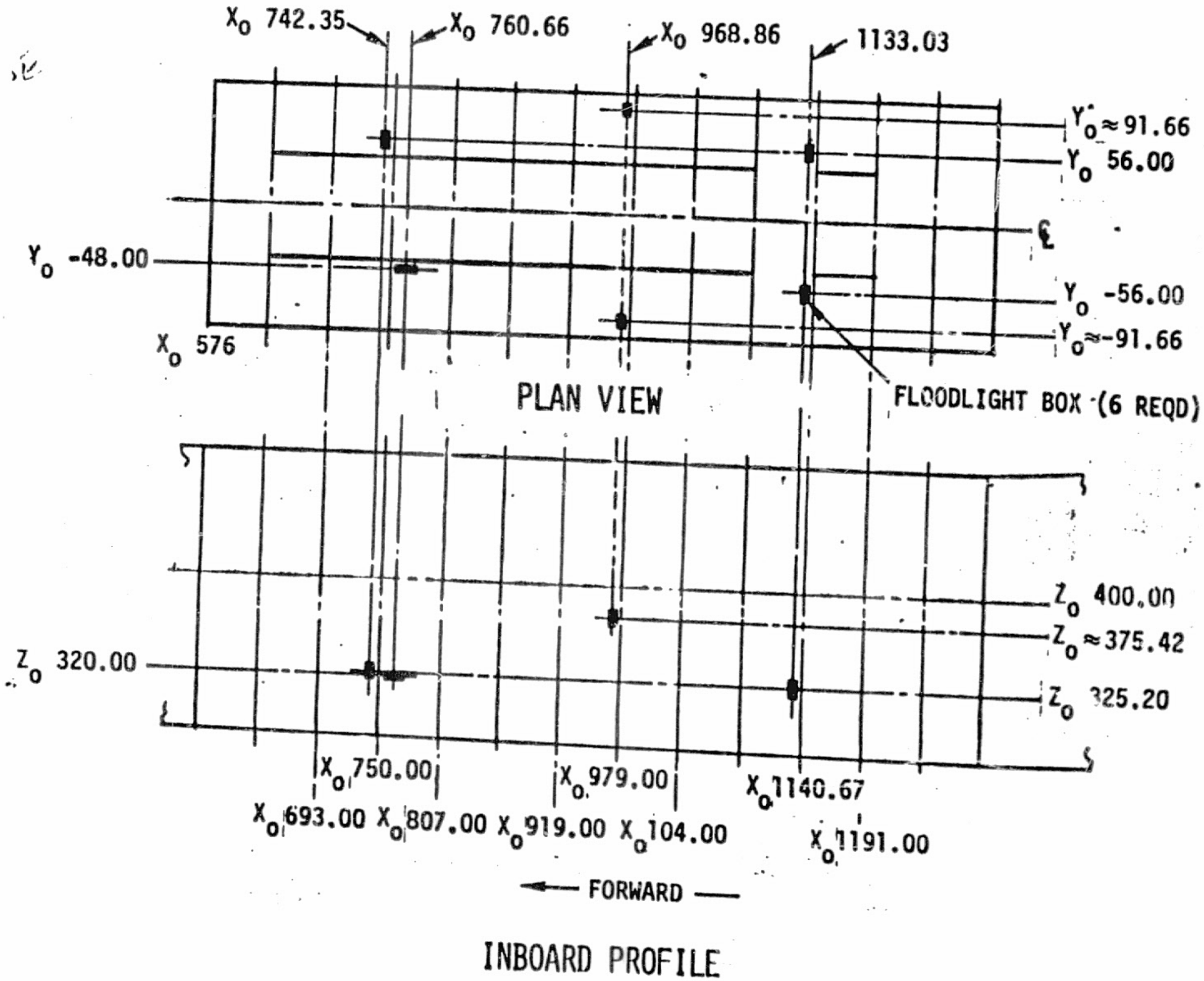


Figure 3.3-8 Coordinates of Orbiter Payload Bay Sidewall Floodlights

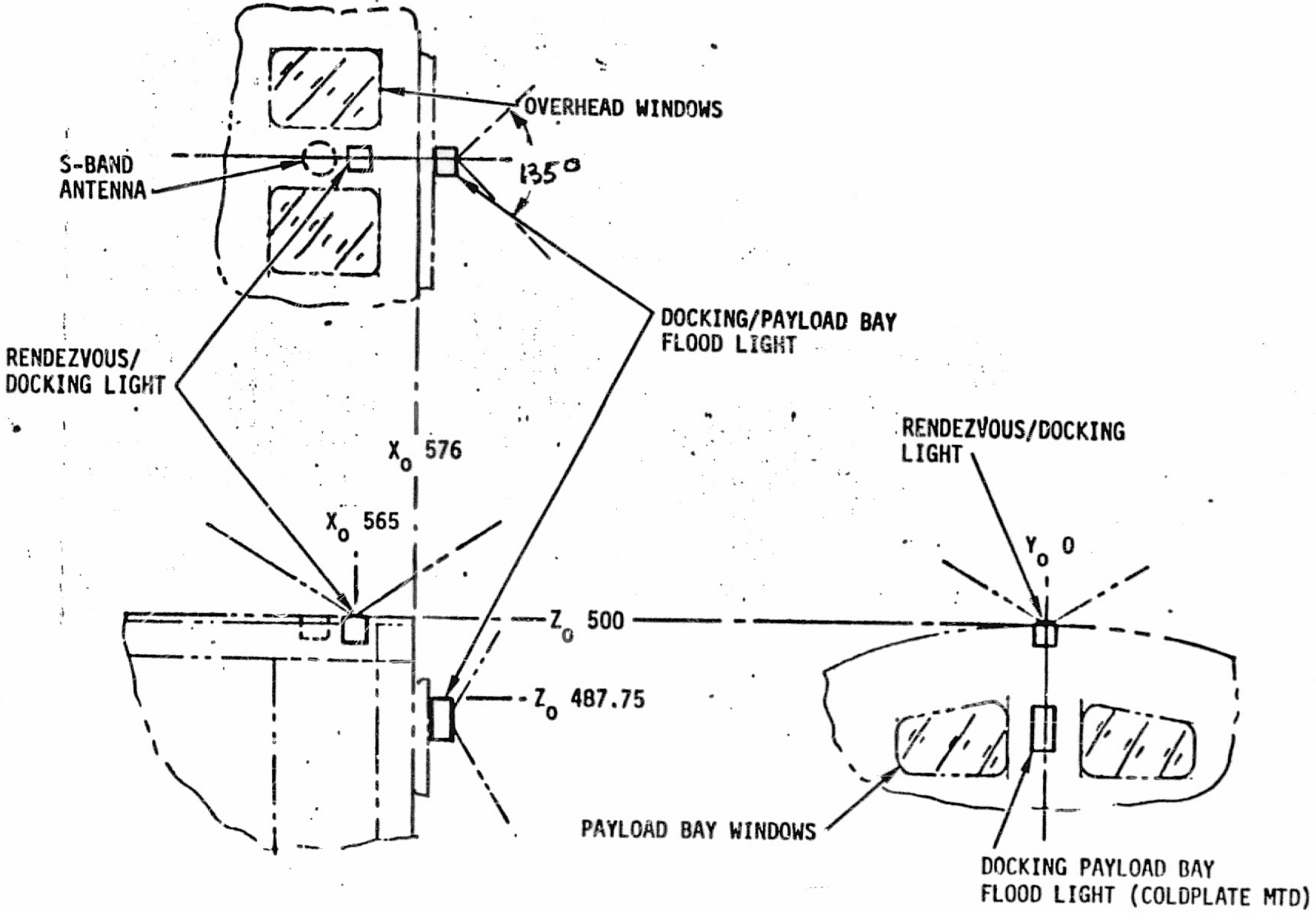


Figure 3.3-9 Coordinates of Orbiter Baseline External Lighting

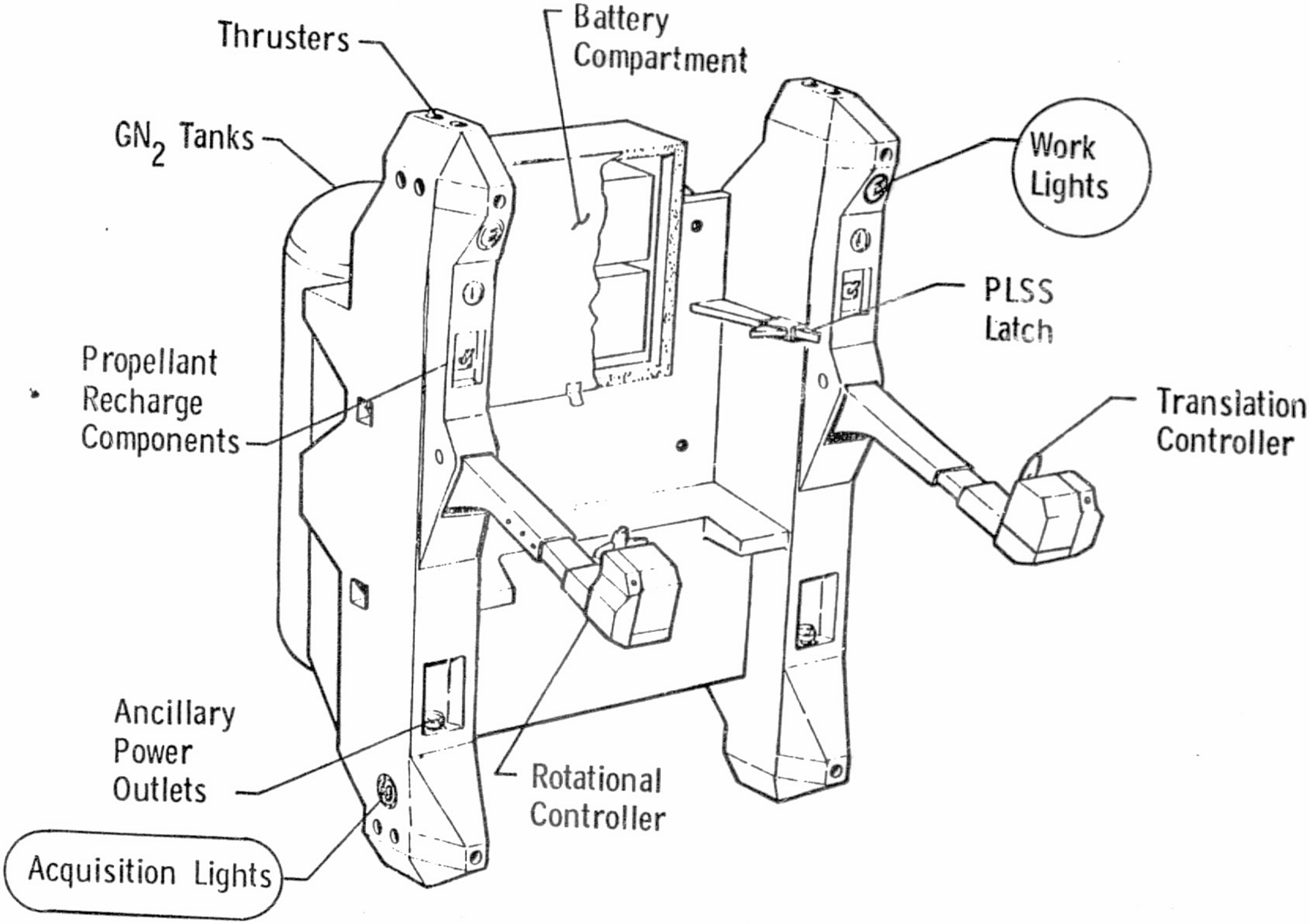


Figure 3.3-10 Manned Maneuvering Unit Lights

Table 3.3-12 MMU Power Consumption Parametrics

• Battery Capacity (full charge, both batteries)	720 watt-hours
• Average Maximum MMU Operational Load, 30 watts x 6 hour mission	- 180 watt-hours
• Power available for orbital operations support:	540 watt-hours
- Floodlights (2) - 25 watts total	
- Power outlets (2) - 28V DC @ 2 amp max each	
• Typical mission - 6 hour duration	
- 2 floodlights operational @ worksite for 5 hrs	125 watt-hours
- camera operational @ 0.5 amp for 2 hours	28 watt-hours
- 1 power tool operational @ 1.5 amp for 5 hours	210 watt-hours
Margin = 177 watt-hours (batteries recharged/replaced prior to next EVA)	363 watt-hours
• Typical mission - 6 hour duration	
- 2 floodlights operational @ worksite for 1 hour	50 watt-hours
- ancillary equipment operational @ 0.5 amp for 3 hours	42 watt-hours
Margin = 448 watt-hours (no battery recharge required prior to next EVA)	92 watt-hours

tion point at specific work sites will permit shuttling off the rate gyros. This will obviate needs to fire thrusters. Additional energy is then available for illumination, within the battery capacity (Reference 6). Development of lights for the MMU is currently being delayed until a decision is made concerning lamps to be attached to the EMU (space suit). In cases where EVA is performed without use of the MMU, another source of work light is desirable. One possible approach is to mount lamps on the EMU back pack, substantially as shown in Figure 2.3-11. This system was evaluated by Rockwell during mockup tests within the orbiter payload bay and found to be useful (Reference 7). Recently received information from NASA's Johnson Space Center indicates that portable lights are being seriously considered for the EMU. One concept is to provide a Velcro mounting on the helmet, which permits detachment of the lamp and use by the astronaut in a hand-held mode. These lamps would require very little power, perhaps as low as 2 watts.

The MMU typically consumes an average of 30 watts, assuming the rate gyros are on constantly. Since the MMU includes floodlights to illuminate a work area and power outlets to operate auxilliary equipment or tools, this typical load could increase dramatically during orbital operations. It should be noted that the rate gyros would normally be turned off when the MMU is at a work site and that, since thrusters would not normally be firing during this period, significant power can be saved below the nominal 30 watt maximum. Thus, the MMU would typically consume much less than the 180 watt-hours shown in Table 3.3-12, and this power saved would be available to operate ancillary equipment. The 540 watt-hours shown available for orbital operation is, therefore, a worst-case estimate.

Another construction aid which could employ built-in lighting to aid construction is the cherry picker, either as open or closed (pressurized cabin) versions. The open cherry picker model concept under study by Grumman incorporates two lamps mounted on stanchions above and behind the crew member to "provide the astronaut with 20 foot candles* of illumination at the worksite" (Reference 8). A pan and tilt mechanism is included for each lamp (Figure 3.3-12). Another area of concern for illumination on the open cherry picker is the control/display panel which must be readable in both sunlight and darkness. The Grumman studies will investigate use of lamps versus flags, edge lighted versus electroluminescent panels, shielded annunciators and a sunscreen. Such concerns may also apply to EVA - operated work stations on construction fixtures where they are utilized.

TV, Lighting and Intra-Project Geometry Considerations

Certain precautions are necessary in using television cameras for aiding space construction activity on the eclipse side of orbit. A major concern is that the light level not be too bright in the observed scene, whether seen as a reflection or as a source. This leads to the following guidelines:

(1) Arrange work such that the TV camera does not look directly at a bright light source, such as a lamp in the orbiter payload bay. Provide temporary or fixed shields if necessary.

*20 foot candles - 215.2 lux

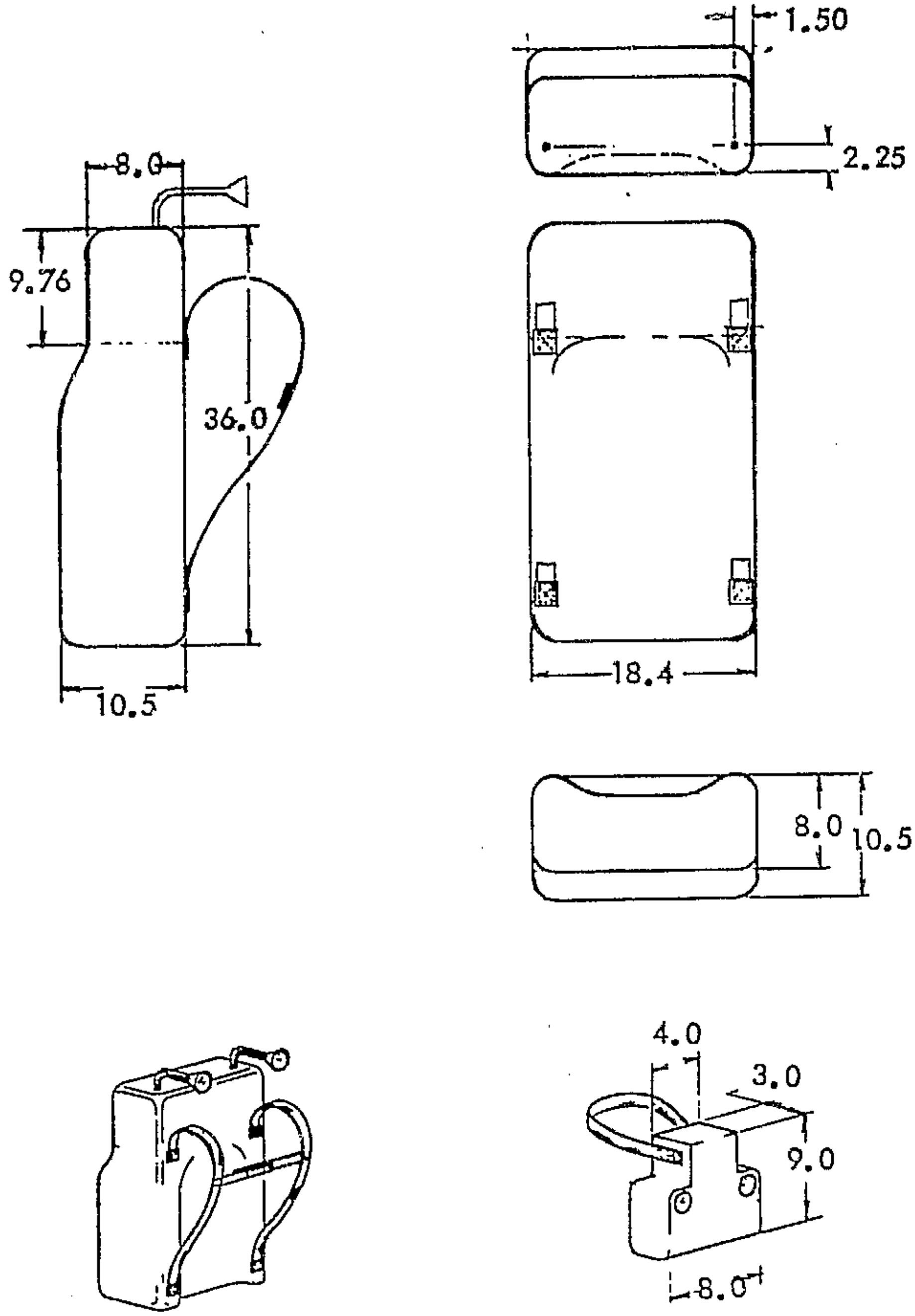


Figure 3.3-11 Work Light Concept for EMU Backpack

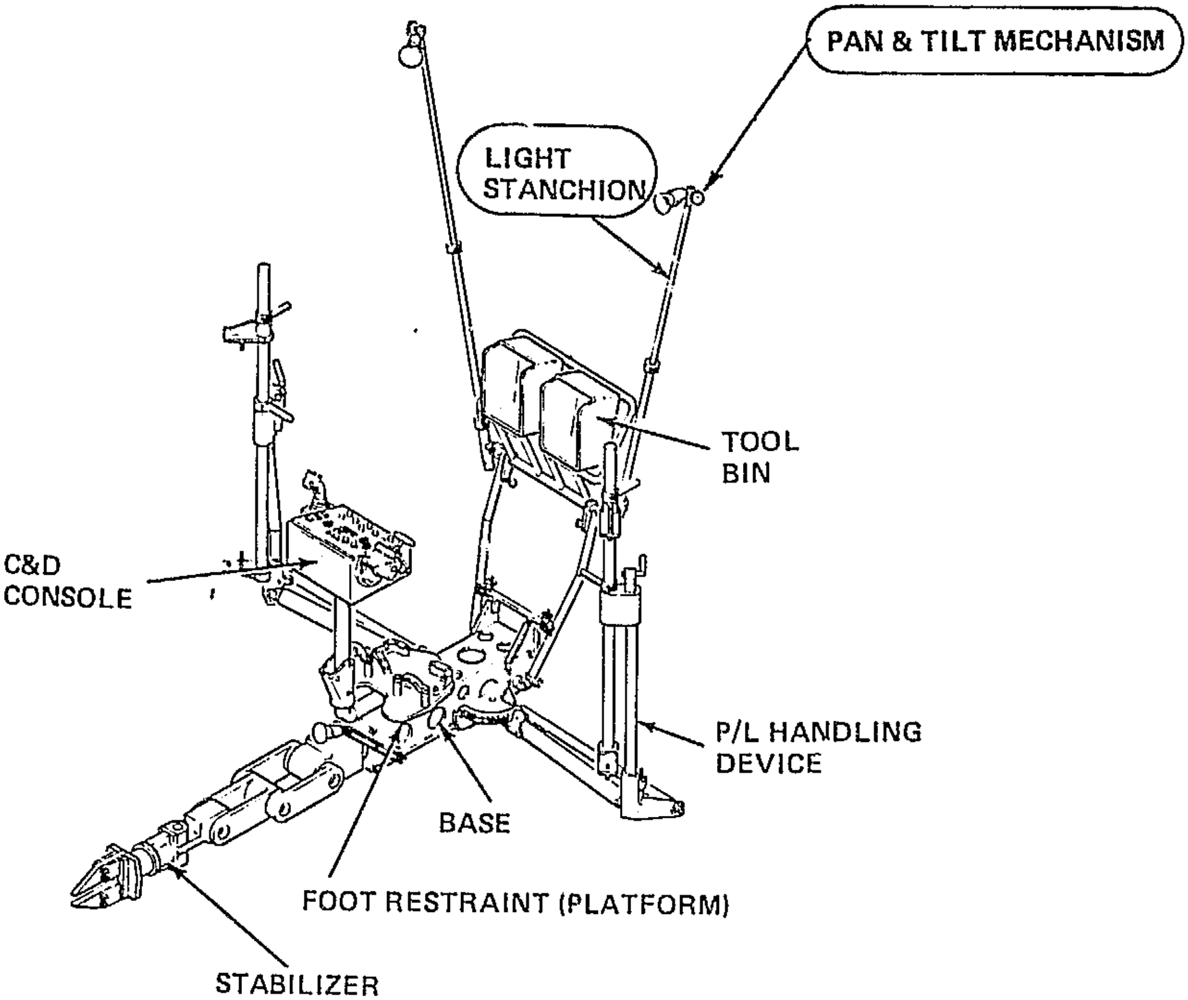


Figure 3.3-12 Open Cherry Picker Development Test Article

(2) Locate lamps to the rear and to the side, above or below TV cameras to avoid direct reflections into the camera lens from planar, specular surfaces perpendicular to the camera axis.

These recommendations have been developed from Rockwell mock-up tests involving EVA and remote viewing during integrated operations with payloads in the orbiter payload bay (References 7 and 9).

Another aspect of interaction with TV, lights and construction geometry relates to the location and orientation with respect to payloads. For example, the standard TV camera and light system mounted near the end effector of the orbiter remote manipulator system is essentially "blind" once it has been engaged to the side of a large module. The camera cannot "see" around or through the module to help the operator guide the RMS motion. However, a tilt and pan system could be provided for the camera and light on the end effector, which may be accurately indexed at 90° pitch or yaw angle from the previous alignment. Thus, one can guide the module toward a target mounted on the construction site and aid installation of the module. Figure 3.3-13 illustrates an example of this concept for grappling a large structure and docking it to an Orbiting Service Module (OSM). Such an approach dictates location and orientation of grasping points on large modules.

Alternately (and perhaps concurrently), there should be provided a set of remotely located TV cameras on the construction fixture, on the module on the construction project itself. These may be pre-positioned by the RMS, EVA, or other means, to aid transport and installation. Such remote cameras are analogous to those provided in the orbiter payload bay to aid payload withdrawal and retrieval. Mockup tests at the NASA/JSC Manipulator Development Facility have demonstrated the value of such aids.

3.3.4 Orientation and Natural Illumination for Sunlit Side of Orbit

Several questions concerning vision and spacecraft orientation on the sunlit side of low earth orbit have been raised during previous space construction studies. For example, General Dynamics analysts recommend against orientations involving viewing conditions in which the earth appears as a brightly lit background to the observed construction activity (Reference 10). Also, it is a well known fact that direct view of the solar disk is harmful to the human eye, and to most TV tubes. Reflection of the sun from highly specular surfaces, such as the radiators on the orbiter payload bay doors, also may be harmful to vision and to TV tubes. The latter is particularly troublesome if the solar rays are concentrated by the curvature of the orbiter thermal radiator surfaces. Finally, there are concerns about glare, high contrast and range of brightness (brightness ratios) between sunlit surfaces and adjacent surfaces in deep shadows, a condition which tends to be accentuated in space. Each of these problem areas was investigated during the analysis effort, and results are reported herein.

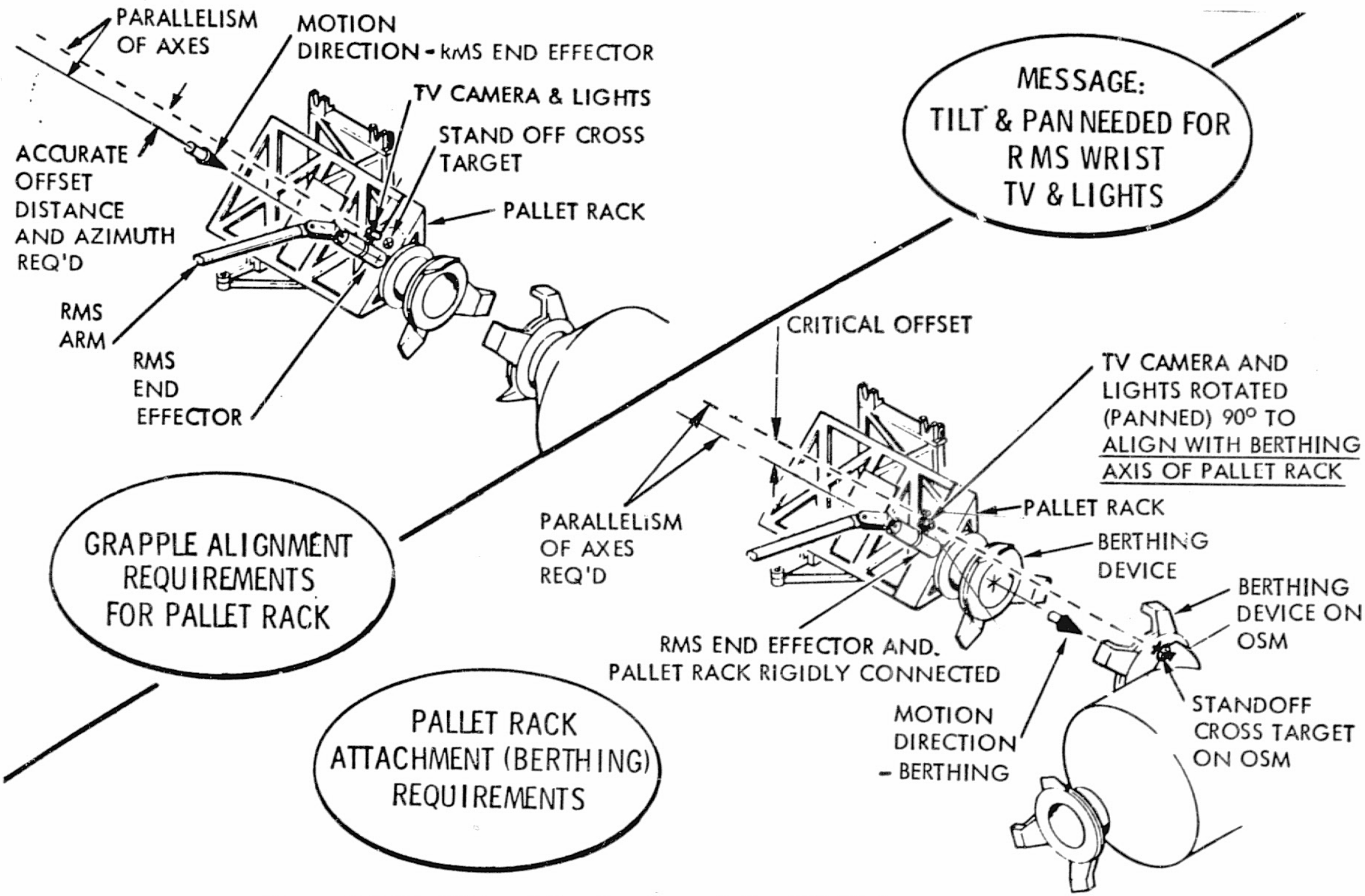


Figure 3.3-13 Tilt and Pan Requirement for RMS TV Camera and Lights to Install/Join Large Objects

High Brightness Background

A large area of cloud cover on the earth can provide a high-brightness background when sunlight is reflected from it. Also, the Shuttle Orbiter has many surfaces with a high reflectivity (ρ = .80 to .95), which could appear as a bright background when observed by a TV camera or EVA crew member looking toward those surfaces. Is this a serious problem? If the surfaces are not specular, the answer is generally "no", for these high-brightness conditions can be reduced to acceptable levels for the human eye by filters of various absorptivities. Table 3.3-13 provides typical transmissability data for various orbiter window, EMU helmet and light filter combinations, as well as the effective reduction in luminance from the sun. For the TV cameras one can use filters, iris diameter control and gain control to reduce undesirable effects of background brightness. In most cases, the more appropriate question is one of the ratio of brightness of the background to the brightness of the observed target (construction project).

Brightness Ratios

In order to see detail contour and marking information on the surfaces of a construction project, it is necessary that the ratio of background brightness to foreground brightness be within the range of capability of the sensors. Whereas the human eye can accommodate to a wide brightness range, TV cameras are usually more limited. Preliminary investigations suggest that TV cameras might accommodate to a range between 30:1 to 50:1. An investigation was performed to evaluate potential problems of brightness contrast in low earth orbit, and calculations were performed to estimate apparent brightness levels and brightness ratios under a wide range of conditions. Initially, information on typical light flux levels in low earth orbit was sought from the literature. Since no flux level measurement data were found, it was necessary to calculate flux levels.

Parenthetically, it appears valuable to initiate a program to measure typical light flux levels at different altitudes and directions in earth orbit. The resulting data would support analyses of future large space construction operations. It appears that more investigation and analysis has been performed in relation to other planets in the solar system than for earth itself.

For purposes of simplifying analyses, it was assumed that all structure would have a coating reflectivity of 0.5. The orbiter spacecraft and construction equipment surface reflectivities were obtained from the best available information and appear in Table 3.3-14. For future reference and analyses, reflectivity data on other selected typical surfaces for large spacecraft were also obtained. These data appear in Figure 3.3-14.

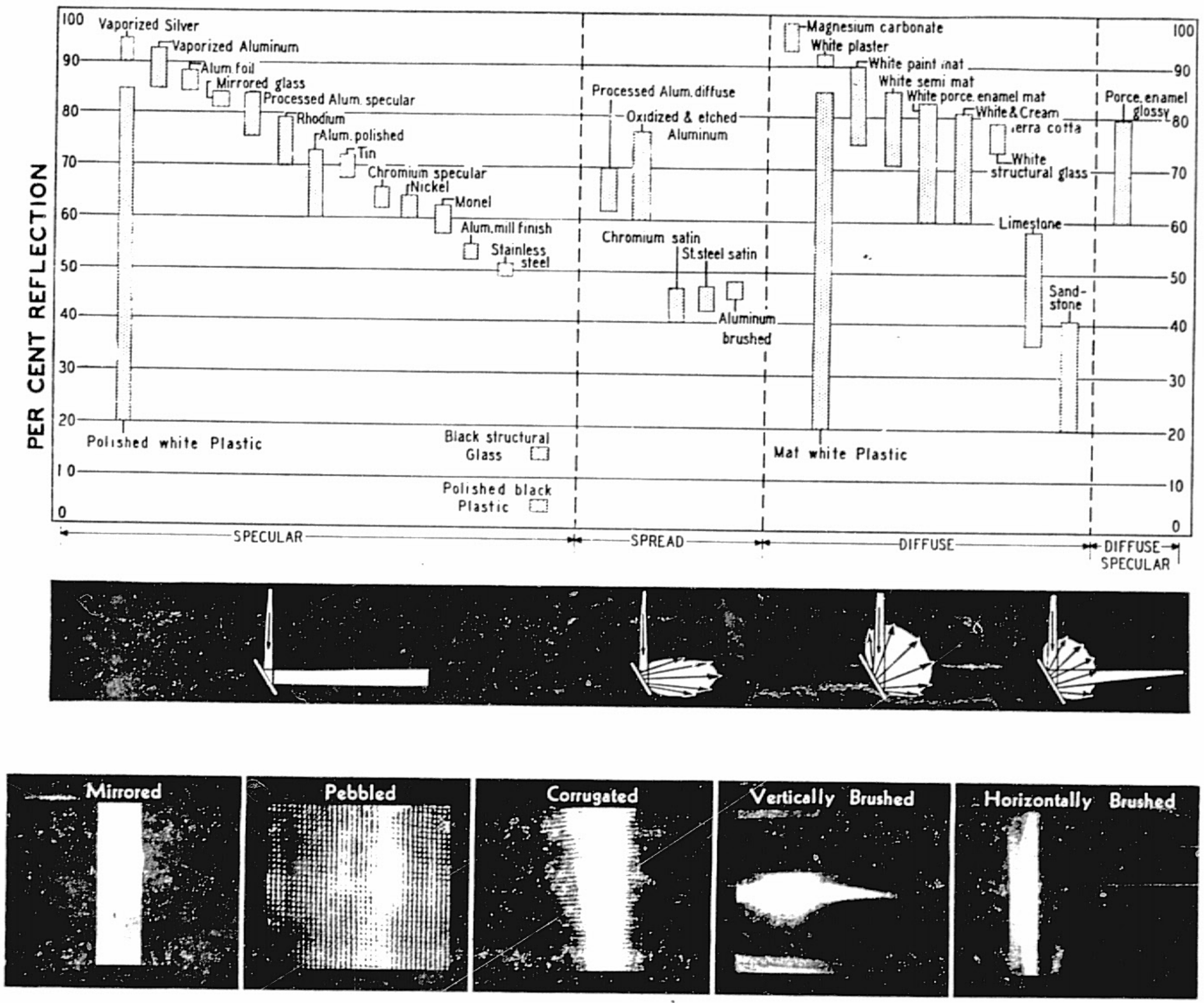
In the space construction environment, many surfaces can act as either a light source (direct or reflected) or as a viewing background as shown previously in Figure 3.3-1. For example, the orbiter payload bay, radiators and wings can act as a source, by reflecting sunlight, earthshine (reflected sunlight from clouds, sea or land), starlight, moonlight, or artificial light

Table 3.3-13
Typical Transparency and Filter Combinations
Transmissibilities and Resulting Apparent Illuminances of
Sunlit Surfaces

No.	Condition	Transparency/Filter Transmissibility Item	%	Overall Effect-Transmissability	Surface Illuminance Lux	Surface Illuminance Ft-Candle
0	Sun-Direct	Space	100%	100%	1.46×10^5	13,500
1	Aft Flight Deck Aft Window View 2 Glass Panes	Outer Pane, Inner Pane		88%	1.28×10^5	11,880
2	2 Glass Panes plus filter	Filter	18%	15.8%	2.3×10^4	2,133
		2 Glass Panes	88%			
3	Aft Flight Deck Overhead View 3 Glass Panes	Outer Pane, Middle Pane, Inner Pane		81%	1.18×10^5	10,935
4	3 Glass Panes plus filter	Filter	18%	14.6%	2.06×10^4	1,968
		3 Glass Panes	81%			
5	EVA Helmet Low Density Filter	Lexan Shell	80%	7.48%	7.00×10^4	6,480
		Lo-D Filter	60%			
6	High Density Filter Low Density Filter EVA Helmet	Hi-D Filter	16+4%	5.8 to 9.6%	8.46×10^3 to 1.40×10^4	783 to 1,296
		Lo-D Filter, Lexan Shell	48%			

Table 3.3-14
Typical Reflection Characteristics for Orbiter
Spacecraft and Construction Equipment

	Reflectance
Cargo Bay Liner Teflon Impregnated Glass Fabric	85 - 90%
Payloads - Painted S13G White	85%
Silver Coated Teflon	95%
Orbiter Radiators	90% (96 - 100% specular)
Orbiter Wings (Upper Surface) and Body	68%
Structural Materials	35 - 85% (Low Specularity Desirable)
Solar Arrays (Cells)	2 - 40%



Plane specular or mirrored surfaces appear dark except when the eye is in the path of the beam. Then the reflected image is only slightly less bright than the source. Notice the patch of highlights produced by the pebbled surface and the spreading of the light by brushed and corrugated surfaces.

Figure 3.3-14 Reflection Characteristics of Materials onto a construction project in process near the orbiter. Also, these orbiter surfaces may be seen as backgrounds to the construction truss work or modules, when viewed by an EVA crew member or a TV camera looking toward the orbiter.

In order to present succinct data on brightness ratios for many of the possible combinations of direct and reflected lighting on structure and on backgrounds, a series of matrix charts were prepared for this systems analysis study. One chart covers cases of a single source of illumination versus a wide range of background conditions (Table 3.3-15). For other charts, combinations of two sources of lighting were shown for a given specified background (such as bright clouds, low brightness earth, orbiter payload bay, orbiter radiator or wing) in Tables 3.3-16 to 3.3-21. In these charts, the same direct and reflected light sources are listed horizontally and vertically in order from left to right and top to bottom respectively. Brightness ratios are listed at intersections of the two conditions. These lists remind us that all natural illumination except starlight comes originally from the sun. However, lighting on a construction project may come directly from the sun or be reflected from sunlight falling on the orbiter or from sunlight which first hits the earth, then is reflected onto the construction work or is reflected onto the various surfaces of the orbiter and then reflected again onto the structure. The large number of potential reflectors creates an extensive analysis problem. However, there are many possible cases which would be either meaningless or trivial, rarely encountered, or known to involve obvious unacceptable viewing conditions. These were noted on the charts by shading or asterisks. Still, it was judged not cost effective to attempt to complete analysis of all possible remaining combinations for purposes of this study project. However, a significant number of important cases were examined to get a sampling of expected conditions.

The results from the approximately 50 cases analyzed showed brightness ratios between structure and background of generally less than 10:1. These analysis results indicate that seeing conditions for construction in low earth orbit are actually often quite favorable. In fact, the multiple diffuse reflections from the wide angle of visible earth surface may provide very good lighting conditions in many cases, particularly by "filling in" the deep shadows which are possible in airless space. The main problems will be work interruption caused by avoiding direct view of the sun (within about 2° of line of sight) or viewing the sun in highly specular reflective surfaces such as the orbiter radiators or glass surfaces on solar arrays at certain unfavorable angles.

Another, more subtle problem, can also occur: that of shifting from viewing a dark structure against a lighter background to viewing a lighter structure against a darker background. At the cross-over time, a lack of contrast may cause problems in discriminating outlines and details of the structure. Again, these conditions are unlikely to occur frequently or to last long, and may be overcome to some degree by color contrast between structure and background, and by distinctive markings on key structural areas where visual alignment is required. However, it is probably valuable to develop methods to predict the frequency and duration of these and other potential work interruption periods related to illumination. In some cases, there may be significant impacts which should be controlled by shading of structure, redesign of construction fixtures or even by the assembly sequence for the project.

Table 3.3-15. Space Construction Natural Illumination Analysis, Single-Source Illuminating Structure (Shuttle Orbiter Base)

STRUCTURE ILLUMINATION SOURCES / VIEWING BACKGROUNDS	SUN																
					EARTH, HI-ρ				EARTH, LO-ρ				MOON				
	DIRECT SUN	PAYLOAD BAY	RADIATOR	WING/BODY	DIRECT	P/L BAY	RADIATOR	WING/BODY	DIRECT	P/L BAY	RADIATOR	WING/BODY	DIRECT	P/L BAY	RADIATOR	WING/BODY	STARS
SUN	NOT ACCEPTABLE																
MOON	SHADOWED AREAS												(d)				
STARFIELD	DIFFICULT TO SEE (b)												(d)				OK
EARTH, HIGH-ρ	1.7		(d)		2.3	3.4	OK		RARE								
EARTH, LOW-ρ	6.2		(d)		RARE				1.7	1.4	OK						
P/L BAY, SUNLIT	1.7	2.5	(d)										NOT ACCEPTABLE (c)				
P/L BAY, E.L., HI-ρ	1.7		(d)		1.7	2.5			RARE								
P/L BAY, E.L., LO-ρ			(d)		RARE				2.0	2.5							
P/L BAY, MOONLIT					NOT APPLICABLE								(d)				
P/L BAY, STARLIT													(d)				
RADIATOR, SUNLIT (a)			(d)														
RADIATOR, E.L., HI-ρ			(d)		1.8				RARE				NOT ACCEPT. (c)				
RADIATOR, E.L., LO-ρ	1.7		(d)		RARE				2.1								
RADIATOR, MOONLIT	NOT APPLICABLE												NOT USEFUL				
RADIATOR, STARLIT													(d)				
WING/BODY, SUNLIT	1.4		(d)							6.2			NOT ACCEPTABLE (c)				
WING/BODY, E.L., HI-ρ			(d)		1.4				RARE								
WING/BODY, E.L., LO-ρ			(d)		RARE												
WING/BODY, MOONLIT	NOT APPLICABLE												(d)				
WING/BODY, STARLIT													(d)				

NOTES: Numerical values are ratios of brightness, higher illumination level divided by lower.

(a) Reflection of solar disc in radiator is not acceptable viewing condition.

(b) Illuminated structure probably much brighter than shadowed areas. Brightness ratios not relevant (additional reflected light desired).

(c) Illuminated background much brighter than structure. Brightness ratios not relevant (additional reflected light on structure desired).

(d) Complex reflection patterns from specular radiator prevents analysis at this level.

Table 3.3-16. Space Construction Natural Illumination Analysis, Double Sources on Structure, Radiator Background Cases (Shuttle Orbiter Base)

STRUCTURAL ILLUMINATION SOURCES / RADIATOR AND STRUCTURE LIGHTING SOURCES		SUN																
						EARTH, HIGH ρ				EARTH, LOW ρ				MOON				
		DIRECT SUN	PAYLOAD BAY	RADIATOR (a)	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	STARS
SUN	DIRECT SUN (b)	●																
SUN	P/L BAY (b)		●															
SUN	RADIATOR (a) (b)			●														
SUN	WING (b)			(f)	●													
SUN, EARTH HIGH ρ	DIRECT (c)	1.8	2.0	(f)	1.4	●	2.7	9.5	3.4	RARE								
SUN, EARTH HIGH ρ	P/L BAY (c)	2.7	2.7	(f)	2.7	2.7	●	9.5	3.4									
SUN, EARTH HIGH ρ	RADIATOR (c)			(f)		9.5	9.5	●	3.4									
SUN, EARTH HIGH ρ	WING (c)	1.8	1.8	(f)	1.8	3.4	3.4	3.4	●									
SUN, EARTH LOW ρ	DIRECT (d)	3.6	2.4	(f)	2.1	RARE				●	2.7	9.5	3.4					
SUN, EARTH LOW ρ	P/L BAY (d)	4.6	3.0	(f)	2.7					2.7	●	9.5	3.4					
SUN, EARTH LOW ρ	RADIATOR (d)			(f)						9.5	9.5	●	3.4					
SUN, EARTH LOW ρ	WING (d)	5.7	3.8	(f)	2.1					3.4	3.4	3.4	●					
SUN, MOON	DIRECT (e)	PRACTICALLY SIMILAR TO SINGLE-SOURCE CASES—NOT APPLICABLE												●			OK	
SUN, MOON	P/L BAY (e)														●			
SUN, MOON	RADIATOR (e)														OK	●		
SUN, MOON	WING (e)															(f)	●	
STARS																(f)		

NOTES: NUMERICAL VALUES ARE RATIOS OF BRIGHTNESS, HIGHER VALUE DIVIDED BY LOWER.

● CASES ARE IMPOSSIBLE OR MEANINGLESS (DUPLICATION OF SOURCES).

(a) REFLECTION OF SOLAR DISC IN RADIATOR IS NOT ACCEPTABLE VIEWING SITUATION.

(b) RADIATOR ILLUMINATED BY DIRECT SUNLIGHT IN THESE CASES (BY ASSUMPTION)

(c) RADIATOR ILLUMINATED BY HIGH-BRIGHTNESS, SUNLIT EARTH IN THESE CASES (BY ASSUMPTION).

(d) RADIATOR ILLUMINATED BY LOW-BRIGHTNESS, SUNLIT EARTH IN THESE CASES (BY ASSUMPTION).

(e) RADIATOR ILLUMINATED BY FULL MOON IN THESE CASES (BY ASSUMPTION).

(f) COMPLEX REFLECTION PATTERN OF POINT SOURCE FROM SPECULAR RADIATOR PREVENTS ANALYSIS AT THIS LEVEL.

Table 3.3-17. Space Construction Natural Illumination Analysis, Double Sources on Structure, Starfield or Dark Earth Background Cases (Shuttle Orbiter Base)

STRUCTURAL ILLUMINATION SOURCES			SUN																
							EARTH, HIGH ρ				EARTH, LOW ρ				MOON				
			DIRECT SUN	PAYLOAD BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	STARS
SUN		DIRECT SUN	●				(DUPLICATES OPPOSITE HALF)												
SUN		P/L BAY		●															
SUN		RADIATOR	(b)	(b)	●														
SUN		WING			(b)	●													
SUN	EARTH HIGH ρ	DIRECT	1.3		(b)		●												
SUN	EARTH HIGH ρ	P/L BAY	1.5	(d)	(b)		1.5	●											
SUN	EARTH HIGH ρ	RADIATOR			(b)				●										
SUN	EARTH HIGH ρ	WING			(b)					●									
SUN	EARTH LOW ρ	DIRECT	3.6		(b)		RARE (a)				●								
SUN	EARTH LOW ρ	P/L BAY			(b)							●							
SUN	EARTH LOW ρ	RADIATOR			(b)								●						
SUN	EARTH LOW ρ	WING			(b)									●					
SUN	MOON	DIRECT	PRACTICALLY SIMILAR TO SINGLE-SOURCE CASES— (c)												●	OK	OK	OK	
SUN	MOON	P/L BAY													OK	●	OK	OK	
SUN	MOON	RADIATOR													OK	OK	●	OK	
SUN	MOON	WING													OK	OK	OK	●	
STARS																			●

NOTES: NUMERICAL VALUES ARE RATIOS OF BRIGHTNESS, HIGHER VALUE DIVIDED BY LOWER.
DARK BACKGROUND NOT CONSIDERED IN CALCULATING BRIGHTNESS RATIOS, ONLY HIGHER BRIGHTNESS SOURCES.

● CASES ARE IMPOSSIBLE OR MEANINGLESS (DUPLICATION OF SOURCES)

(a) PRESENCE OF LOW-BRIGHTNESS BACKGROUND USUALLY IMPLIES LOW-BRIGHTNESS LIGHTING ON ALL ORBITER SRUFACES.

(b) COMPLEX REFLECTION PATTERN FROM SPECULAR RADIATOR PREVENTS ANALYSIS AT THIS LEVEL.

(c) LUNAR LIGHT ESSENTIALLY INVISIBLE IN PRESENCE OF BRIGHT SUNLIGHT REFLECTIONS. BRIGHTNESS RATIOS GREATER THAN 10^4.

(d) UNLIKELY BOTH SOURCES CAN ILLUMINATE STRUCTURE DUE TO CONFIGURATION AND EARTH-SUN RELATIONSHIP.

Table 3.3-18. Space Construction Natural Illumination Analysis, Double Sources on Structure Sunlit Bright Cloud Background Cases (Shuttle Orbiter Base)

STRUCTURAL ILLUMINATION SOURCES		SUN: DIRECT SUN	SUN: PAYLOAD BAY	SUN: RADIATOR	SUN: WING	SUN, EARTH, HIGH ρ: DIRECT	SUN, EARTH, HIGH ρ: P/L BAY	SUN, EARTH, HIGH ρ: RADIATOR	SUN, EARTH, HIGH ρ: WING	SUN, EARTH, LOW ρ: DIRECT	SUN, EARTH, LOW ρ: P/L BAY	SUN, EARTH, LOW ρ: RADIATOR	SUN, EARTH, LOW ρ: WING	SUN, MOON: DIRECT	SUN, MOON: P/L BAY	SUN, MOON: RADIATOR	SUN, MOON: WING	STARS
SUN	DIRECT SUN	●								(DUPLICATES OPPOSITE HALF)								
SUN	P/L BAY	2.6	●															
SUN	RADIATOR	(b)	(b)	●														
SUN	WING	1.7	2.7	(b)	●													
SUN, EARTH HIGH ρ	DIRECT	2.3	3.4	(b)		●												
SUN, EARTH HIGH ρ	P/L BAY	3.4	(d)	(b)		3.4	●											
SUN, EARTH HIGH ρ	RADIATOR	2.3	2.6	(b)				●										
SUN, EARTH HIGH ρ	WING			(b)					●									
SUN, EARTH LOW ρ	DIRECT					RARE (a)				●								
SUN, EARTH LOW ρ	P/L BAY										●							
SUN, EARTH LOW ρ	RADIATOR											●						
SUN, EARTH LOW ρ	WING												●					
SUN, MOON	DIRECT					PRACTICALLY SIMILAR TO SINGLE-SOURCE CASES (c)								●				
SUN, MOON	P/L BAY														●			
SUN, MOON	RADIATOR															●		
SUN, MOON	WING																●	
STARS																		●

NOTE: NUMERICAL VALUES ARE RATIOS OF BRIGHTNESS, HIGHER VALUE DIVIDED BY LOWER.

● CASES ARE IMPOSSIBLE OR MEANINGLESS (DUPLICATION OF SOURCES)

(a) PRESENCE OF HIGH-BRIGHTNESS BACKGROUND USUALLY IMPLIES HIGH-BRIGHTNESS LIGHTING OF ALL ORBITER SURFACES.

(b) COMPLEX REFLECTION PATTERN OF POINT SOURCE FROM SPECULAR RADIATOR PREVENTS ANALYSIS AT THIS LEVEL.

(c) LUNAR LIGHT ESSENTIALLY INVISIBLE IN PRESENCE OF BRIGHT SUNLIGHT REFLECTIONS. BRIGHTNESS RATIOS GREATER THAN 10^5.

(d) UNLIKELY THAT BOTH SOURCES CAN ILLUMINATE STRUCTURE DUE TO CONFIGURATION OR EARTH-SUN RELATIONSHIPS.

Satellite Systems Division
Space Systems Group
Rockwell International

Table 3.3-19. Space Construction Natural Illumination Analysis, Double Sources on Structure, Low-Brightness Earth Background (Sunlit) Cases (Shuttle Orbiter Base)

STRUCTURAL ILLUMINATION SOURCES		SUN																
						EARTH, HIGH ρ				EARTH, LOW ρ				MOON				
		DIRECT SUN	PAYLOAD BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	STARS
SUN	DIRECT SUN	●				(DUPLICATES OPPOSITE HALF)												
	P/L BAY	6.2	●															
	RADIATOR	(b)	(b)	●														
	WING			(b)	●													
EARTH HIGH ρ	DIRECT	RARE (a)				●												
	P/L BAY						●											
	RADIATOR							●										
	WING								●									
EARTH LOW ρ	DIRECT	6.2		(b)		RARE (a)				●								
	P/L BAY	6.2	(d)	(b)							●							
	RADIATOR			(b)								●						
	WING			(b)									●					
MOON	DIRECT	PRACTICALLY SIMILAR TO SINGLE-SOURCE CASES— (c)												●				
	P/L BAY														●			
	RADIATOR															●		
	WING																●	
STARS																		●

NOTES: NUMERICAL VALUES ARE RATIOS OF BRIGHTNESS, HIGHER ILLUMINATION LEVEL DIVIDED BY LOWER.

● CASES ARE IMPOSSIBLE OR MEANINGLESS (DUPLICATION OF SOURCES)

(a) PRESENCE OF LOW-BRIGHTNESS BACKGROUND USUALLY IMPLIES LOW-BRIGHTNESS LIGHTING ON ALL SURFACES.

(b) COMPLEX REFLECTION PATTERN OF POINT SOURCE FROM SPECULAR RADIATOR PREVENTS ANALYSIS AT THIS LEVEL.

(c) LUNAR LIGHT ESSENTIALLY INVISIBLE IN PRESENCE OF BRIGHT SUNLIGHT REFLECTIONS.

(d) UNLIKELY THAT BOTH SOURCES CAN ILLUMINATE STRUCTURE DUE TO CONFIGURATION OR EARTH-SUN RELATIONSHIP.

Satellite Systems Division
Space Systems Group
Rockwell International

Table 3.3-20. Space Construction Natural Illumination Analysis, Double Sources on Structure Payload Bay Background Cases (Shuttle Orbiter Base)

PAYLOAD BAY AND STRUCTURE LIGHTING SOURCES \ STRUCTURAL ILLUMINATION SOURCES			SUN																
							EARTH, HIGH ρ				EARTH, LOW ρ				MOON				
			DIRECT SUN	PAYLOAD BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	STARS
SUN		DIRECT SUN	●				(DUPLICATES OPPOSITE HALF)												
SUN		P/L BAY	2.6	●															
SUN		RADIATOR	(b)	(b)	●														
SUN		WING			(b)	●													
SUN	EARTH HIGH ρ	DIRECT	2.3	(e)	(b)	(e)	●												
SUN	EARTH HIGH ρ	P/L BAY		(d)	(b)	(e)		●											
SUN	EARTH HIGH ρ	RADIATOR		(e)	(b)	(e)			●										
SUN	EARTH HIGH ρ	WING		(e)	(b)	(e)				●									
SUN	EARTH LOW ρ	DIRECT		(e)	(b)	(e)	RARE (a)				●								
SUN	EARTH LOW ρ	P/L BAY		(d)	(b)	(e)						●							
SUN	EARTH LOW ρ	RADIATOR		(e)	(b)	(e)							●						
SUN	EARTH LOW ρ	WING		(e)	(b)	(e)								●					
SUN	MOON	DIRECT	PRACTICALLY SIMILAR TO SINGLE-SOURCE CASES— (c)												●				
SUN	MOON	P/L BAY														●			
SUN	MOON	RADIATOR													OK		●		
SUN	MOON	WING																●	
STARS																			

NOTES: NUMERICAL VALUES ARE RATIOS OF BRIGHTNESS, HIGHER ILLUMINATION LEVEL DIVIDED BY LOWER.

● CASES ARE IMPOSSIBLE OR MEANINGLESS (DUPLICATION OF SOURCES).

(a) PRESENCE OF LOW-EARTH BRIGHTNESS ILLUMINATION USUALLY IMPLIES LOWER LEVELS OF ILLUM. ON ALL SURFACES.

(b) COMPLEX REFLECTION PATTERN OF POINT SOURCE FROM SPECULAR RADIATOR PREVENTS ANALYSIS AT THIS LEVEL.

(c) LUNAR LIGHT ESSENTIALLY INVISIBLE IN PRESENCE OF SUNLIGHT REFLECTIONS.

(d) UNLIKELY THAT BOTH SOURCES CAN ILLUMINATE STRUCTURE.

(e) UNLIKELY COMBINATION OF ILLUMINATION SOURCES AND VIEWING CONDITIONS.

Table 3.3-21. Space Construction Natural Illumination Analysis, Double Sources on Structure, Wing/Body Surface Background (Shuttle Orbiter Base)

WING-BODY AND STRUCTURE LIGHTING SOURCES \ STRUCTURAL ILLUMINATION SOURCES			SUN																
							EARTH, HIGH ρ				EARTH, LOW ρ				MOON				
			DIRECT SUN	PAYLOAD BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	DIRECT	P/L BAY	RADIATOR	WING	STARS
SUN		DIRECT SUN	●				(DUPLICATES OPPOSITE HALF)												
		P/L BAY	1.4	●															
		RADIATOR	(b)	(b)	●														
		WING			(b)	●													
	EARTH HIGH ρ	DIRECT					●												
		P/L BAY		(d)	(b)		2.1	●											
		RADIATOR			(b)				●										
		WING			(b)					●									
	EARTH LOW ρ	DIRECT	3.0		(b)		RARE (a)				●								
		P/L BAY		(d)	(b)							●							
		RADIATOR			(b)								●						
		WING			(b)									●					
	MOON	DIRECT	PRACTICALLY SIMILAR TO SINGLE-SOURCE CASES— (c)												●				
		P/L BAY														●			
		RADIATOR															●		
		WING																●	
STARS																			●

NOTES: NUMERICAL VALUES ARE RATIOS OF BRIGHTNESS, HIGHER ILLUMINATION LEVEL DIVIDED BY LOWER.
CASES ARE IMPOSSIBLE OR MEANINGLESS (DUPLICATION OF SOURCES)

(a) PRESENCE OF LOW-EARTH BRIGHTNESS ILLUMINATION USUALLY IMPLIES LOWER LIGHTING LEVELS ILLUMINATING ALL SURFACES.

(b) COMPLEX REFLECTION PATTERNS OF POINT SOURCE FROM SPECULAR RADIATOR PREVENTS ANALYSIS AT THIS LEVEL.

(c) LUNAR LIGHT ESSENTIALLY INVISIBLE IN PRESENCE OF SUNLIGHT REFLECTIONS.

(d) UNLIKELY THAT BOTH SOURCES CAN ILLUMINATE STRUCTURE.

In summary, the foregoing problems may be anticipated in some degree and avoided in the following ways:

(1) Multiple TV camera locations providing a selection of viewing angles.

(2) Shifting the viewing location (for EVA crew).

(3) Providing localized shade panels for diffusers.

(4) Work scheduling.

(5) Construction project re-orientation.

The last listed option has been noted as generally undesirable because of stress loads and fuel economies. It is probably not necessary or cost effective in most cases. The constantly changing orbital conditions will tend to revise viewing angles in relation to the solar arrays within a short time, especially if there is no attempt to maintain a solar inertial attitude control.

Attitude Orientation Considerations

Foregoing considerations of vehicle and construction work orientations have dealt mainly with general problems of light levels and specularity associated with differing orientations rather than specific angular limits and specific tasks involving illumination and viewing with TV. It is now appropriate to examine the geometric interaction of some specific construction activity viewing requirements and the critical combinations of light directions impinging on the construction. Previous discussion has shown that the direction of the sun's rays is the single most critical consideration affecting work interruption on the sunlit side of orbit. A method is needed to analyze and summarize typical viewing directions for a specific project and relate them to probable directions of the sun's rays, based upon stability analyses of the orbiting construction projects. The method used here for the summary presentation of viewing angles is the single-point globographic diagram. In such a presentation it is assumed that all viewing directions originate from a common point in space. All of the critical viewing angles are plotted as points or areas on a sphere surrounding the common point, in terms of angles of longitude and latitude. With respect to the sun's nearly parallel rays of light, such an approximation is quite reasonable as a summary method. Of course, in the actual physical situation there may be a portion of the orbiter, construction project or space base between the observer and the sun, such that there is no problem of viewing the solar disc by an observer. Such obstructions can also be plotted for a specific eye or camera position.

To develop a meaningful example which illustrates the method and presentation, a brief analysis was performed on the space fabricated tri-beam advanced communication platform project, which is shown in Figure 3.3-15. The perspective view in the lower right of the figure indicates (by arrows) the principal viewing directions involved in relation to a proposed construction fixture to be used in conjunction with the space Shuttle Orbiter.

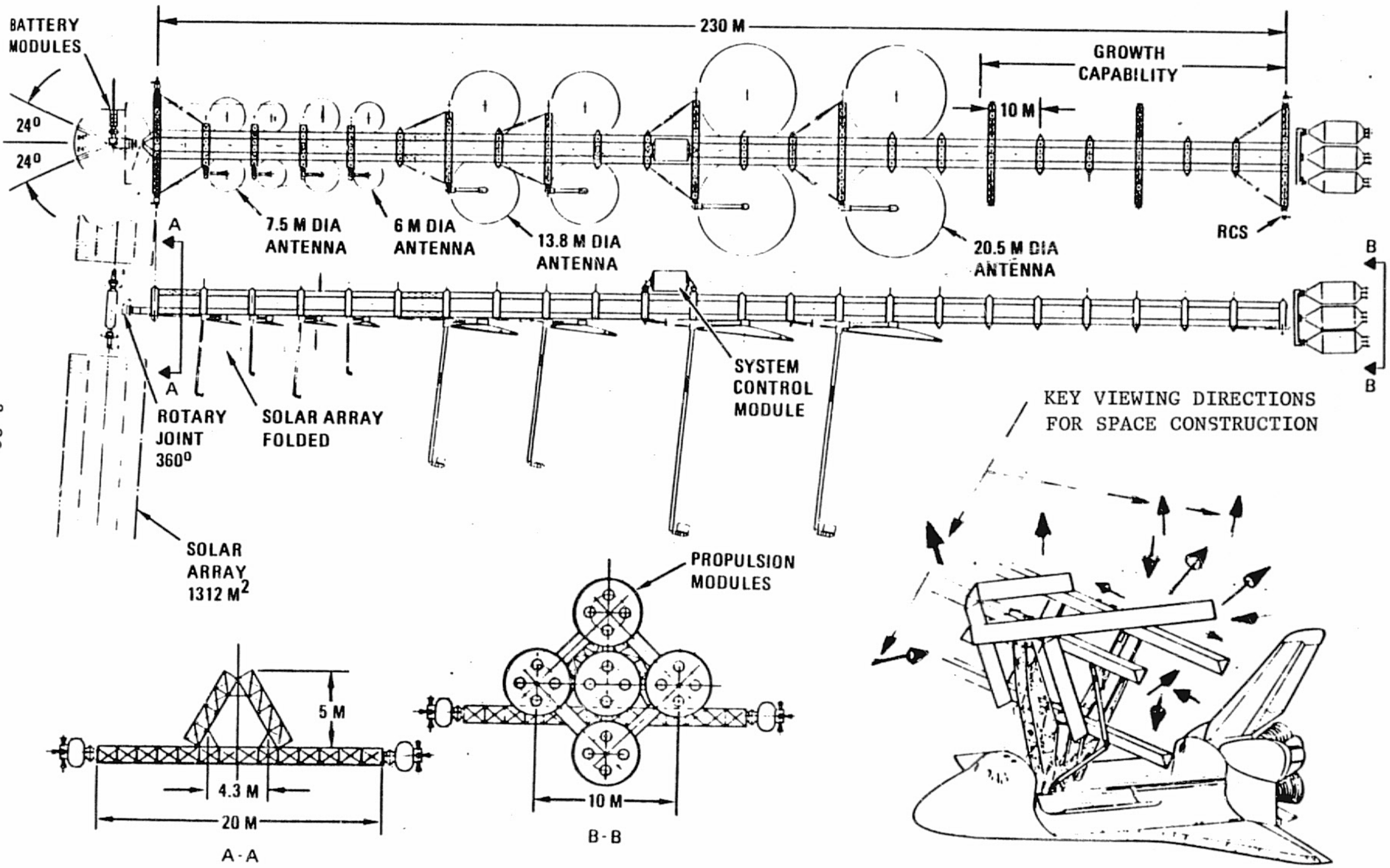


Figure 3.3-15 Space Fabricated Tri-Beam Advanced Communications Platform

Figure 3.3-16 provides supporting illustrations of several key construction operations and probable viewing directions (indicated by arrows and TV camera/ light sets) which were identified and tabulated during the analysis.

Two views of the globographic summary of viewing angles for this project are shown in Figure 3.3-17. The views are a rear quarter view and front quarter view with respect to the orbiter. On the theoretical globe graduated in degrees elevation and azimuth, 0^o of elevation and azimuth correspond to the orbiter's -X axis. Areas (spots) marked with shading slanted upward to the right represent an estimated range ($\pm 5^o$) about the nominal key viewing angles required for TV cameras or EVA crew during construction processes. The directions indicated by arrows shown in Figures 3.3-15 and 3.3-16 would intersect the globe at the center of these cross hatched areas. The dashed lines which surround the shaded areas suggest a wider range of angles where sunlight impingement on a TV camera lens would be undesireable, and where some special shielding may be required to reduce viewing angles. Areas shaded with lines slanting upward to the left, covering the top and a portion of the rear of the globe around the orbiter, are also shown. These indicate probable required viewing angles from the windows of the crew cabin on-orbit stations and the payload bay TV cameras. Such angles would be used for erection operations of the construction fixture, general observations of construction processes, and occasionally for selected critical transport and assembly operations.

Knowing these probable viewing angles, one can consider selection of initial orientations and launch parameters for favorable average viewing conditions, especially where some orientation control is achievable by gravity gradient methods. The question of desirable orientations and possible methods for controlling orientations of space construction is still open, with many unknowns. As an introduction to the subject, some of the more obvious orientations options were identified and briefly evaluated. The orientations considered and resulting effects from natural illumination and thermal effects are summarized in Table 3.3-22. It appears that the solar-fixed inertial attitude gives the best control of natural illumination, and is desirable if a favorable attitude can be maintained. However, any unfavorable aspects would also be maintained continuously throughout the sunlit side of the orbit. In contrast, the earth-oriented and the drifting modes have the potential advantage (as well as problems) of variety; any problem which turns up will likely pass away shortly afterward. More important and specific to the selection of orientations are the problems previously mentioned: fuel requirements and power to maintain positive attitude control, potential loads on the structure during the vulnerable period of construction, and the added complexity and cost of stabilization and control systems which are designed only for the construction period. The free-drift, semi-gravity gradient stabilized orientations tend to reduce such fuel costs and complexities to a minimum.

Based on the foregoing analyses, the following general guidelines were derived for favorable construction orientation selection in relation to the sunlit side of earth orbit:

- o Orient the payload bay toward the earth or toward the horizon such that at least half the earth disc can reflect diffuse sunlight reflections from earth toward the structure and also can be reflected

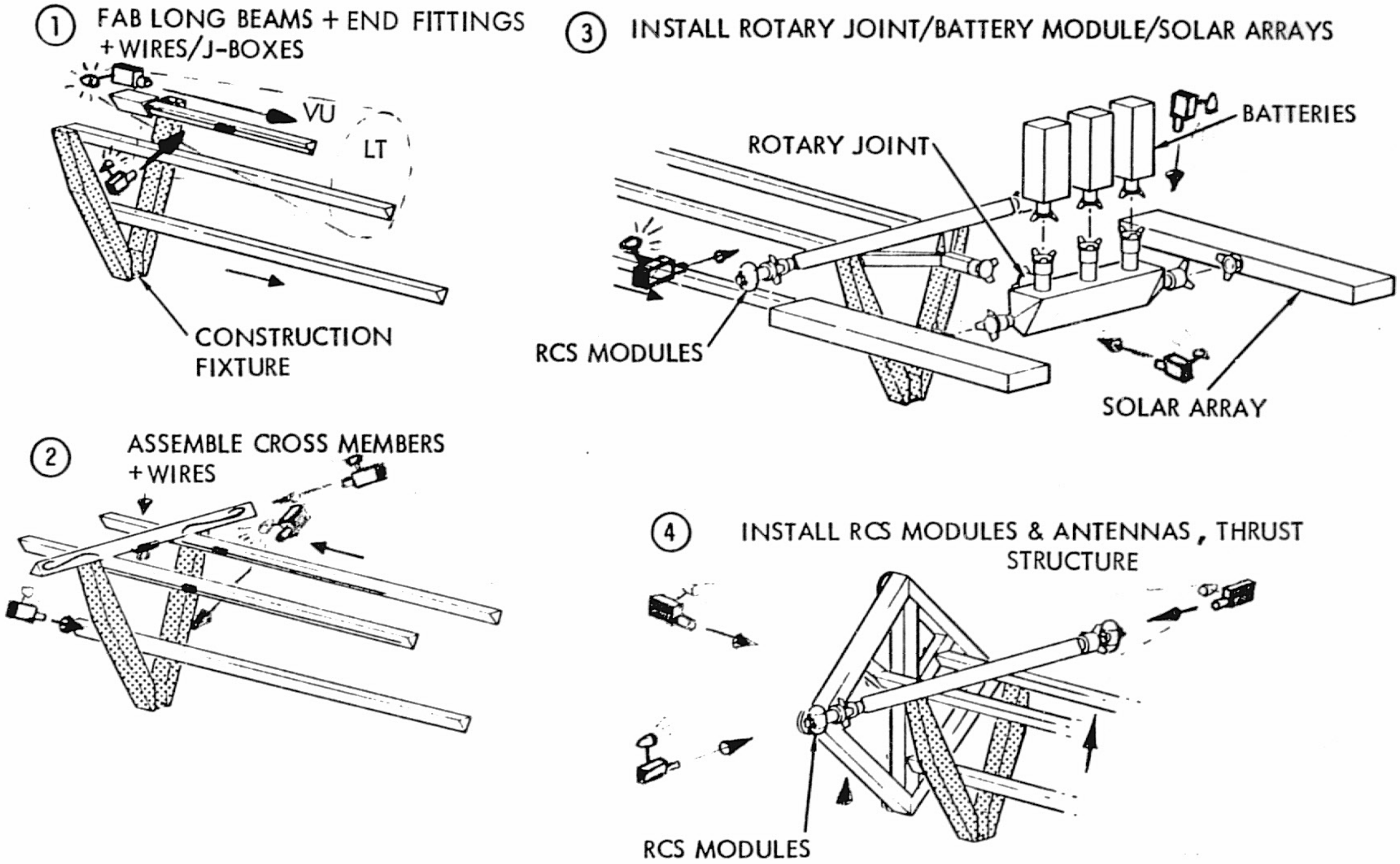


Figure 3.3-16 Key Construction Processes and Associated Viewing Directions

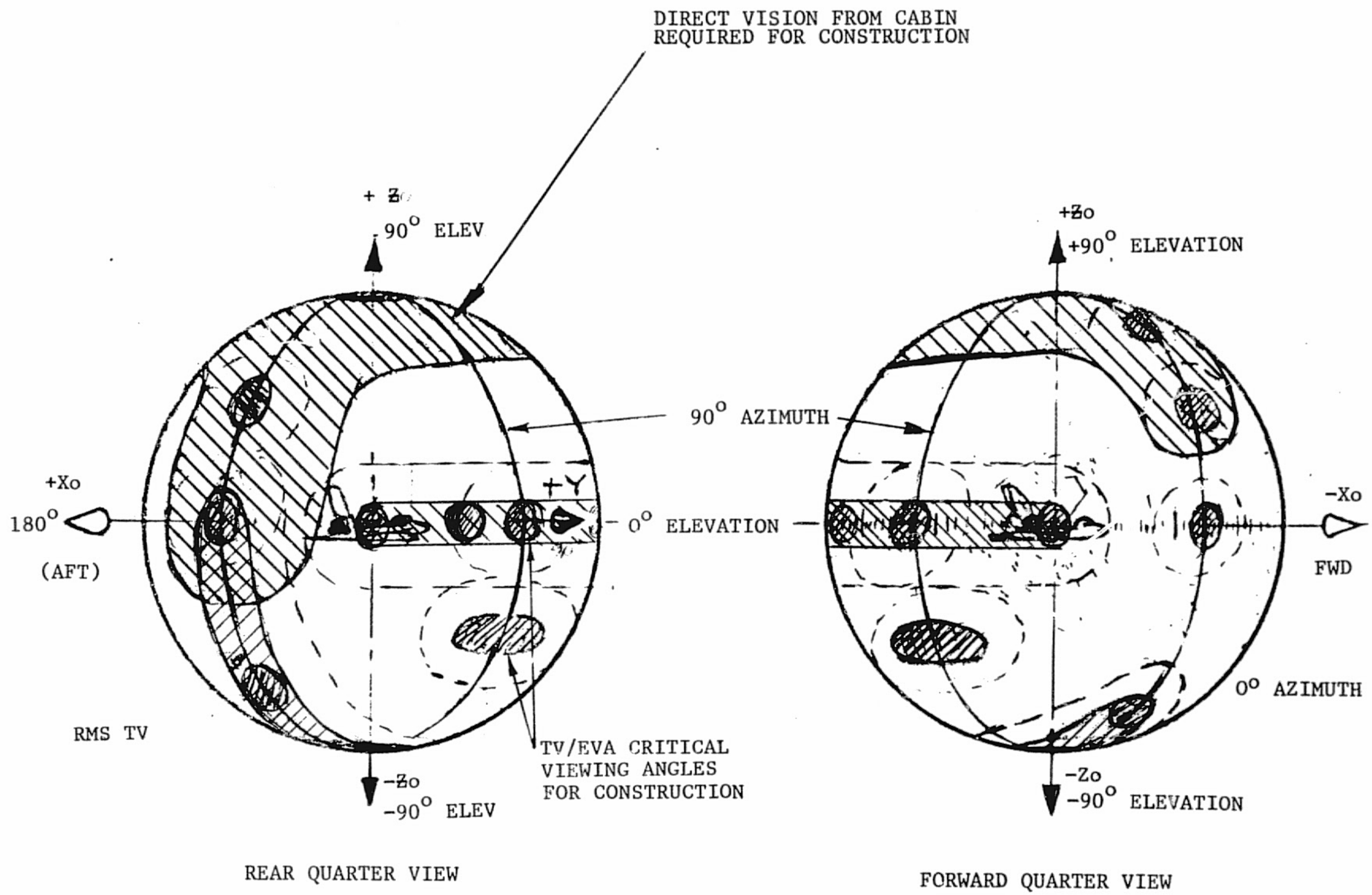


Figure 3.3-17 Globographic Viewing Angle Summary
Space Fabricated Tri-Beam Communications Platform

TABLE 3.3-22

General Considerations for Selecting Space Construction Orientation With Respect to Shuttle Orbiter

ATTITUDE	CONSIDERATIONS FOR VIEWING AND THERMAL RADIATION
Solar-fixed inertial:	
o Nose toward sun	Construction is well illuminated if above cargo bay or to the side of Orbiter. No problems of veiling luminance from sun, good contrast with space, not so good for earth-cloud cover. Favorable for thermal radiation.
o Nose toward sun pitch down $45^{\circ} \pm 10^{\circ}$	Construction is well illuminated by direct sunlight and wing/body reflections. Some bright reflections from Payload bay, minimal from radiators. Reduced glare from earth cloud cover, only from limb, low area, low reflectance. No Solar veiling luminance.
o Nose 90° from sun, wings in alignment with sun rays (across ship)	One side of assembly fixture in solar shadow, but strong reflection from earth on opposite side helps, especially during middle of sunlit period. Favorable for thermal radiation.
Earth-fixed ralationship	
o Wings toward nadir Nose toward pole (South or North as function of time of year, time of launch)	o General Dynamics (SCAFEDS) considers favorable to limit earth viewing and sun viewing. Some problems with sun background when coming around limb of earth. Configurations for requiring over head viewing have part-time viewing directly at sun. Solar shadows over payload bay mitigated by earth reflections when belly of orbiter facing sun. Variable thermal radiation effects.
o Payload bay toward earth	o Bright solar reflections off payload bay at dawn and dusk of orbit. Earth background at all times, very bright at orbital noon, but in shadow of sun; good viewing. Unfavorable for thermal radiation.

TABLE 3.3-22 (Continued)

ATTITUDE	CONSIDERATIONS FOR VIEWING AND THERMAL RADIATION
Earth-fixed relationship (Cont.)	
o Payload bay away from earth	o Very bright reflections from sun in payload bay. Sun background overhead unacceptable during large part of orbit, unfavorable for TV & EVA looking toward orbiter radiators.
Random - free drift	
o Rotating mode - undamped	o Wide variety of angles are not predictable. Optimum approach for vision would be to erect a diffuser/reflector curtain over the payload bay prior to starting construction work, provide artificial illumination. However, radiator heat rejection is adversely affected.
o Gravity gradiant stabilization only. Limited oscillations	o Attitudes under these conditions must be calculated. Shade or diffuser may be desired if attitude unfavorable. Location is critical to radiator heat rejection. Some of earth-fixed relationships may be applicable, depending on configuration, construction strategy.

by the orbiter surfaces. This attitude range tends to avoid solar reflections from the orbiter radiators.

- o Alternatively, orient the orbiter and construction work so as to reflect sunlight from the wings, radiator and payload bay toward the construction work, but from behind the majority of TV cameras and EVA work stations.

- o When possible, orient the orbiter and/or construction base (if used) so as to shield viewing sites from direct solar impingement when work is going on at the work site.

The latter guideline suggests that the sun light should illuminate the belly of the orbiter rather than the payload bay and radiators. Applications of these guidelines are illustrated by a specific example. A preliminary stability analysis of the tri-beam, space fabricated, advanced communication platform during the construction process has indicated the probable orientation conditions, illustrated in Figure 3.3-18, and further explained as follows:

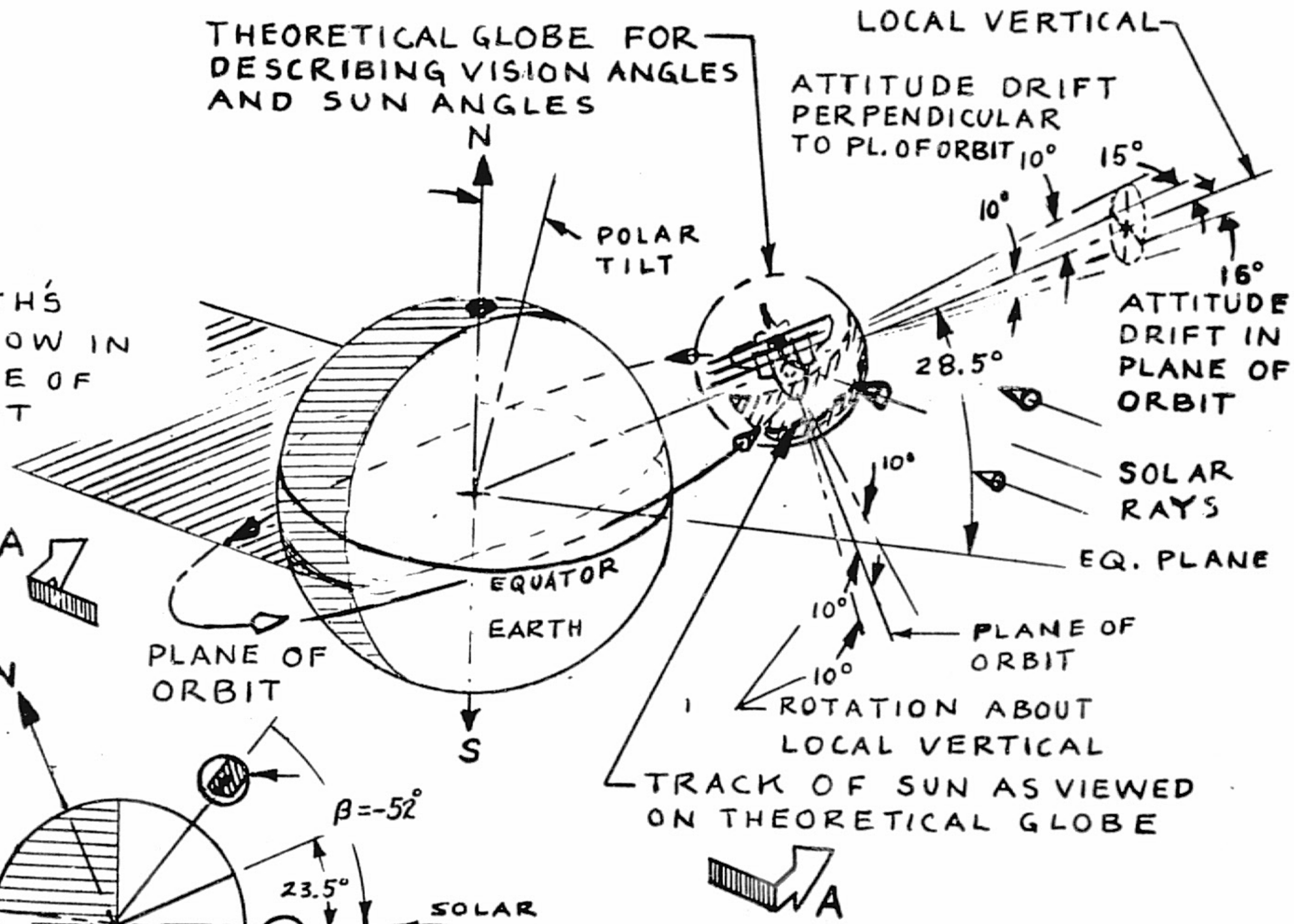
(1) The longitudinal axis of the project (orbiter Y axis) will point toward the center of the earth (along the local vertical) and will drift approximately $\pm 15^{\circ}$, leading or lagging the orbiter, in the plane of orbit. The period is about 1.7 cycles per orbit. The orbiter wings will approximately lie in the plane of the orbit.

(2) The construction project will oscillate approximately $\pm 10^{\circ}$ about the local vertical in directions perpendicular to the plane of the orbit.

(3) The orbiter should be oriented with tail leading (but nose leading is acceptable). It will tend to rotate about the orbiter Y axis, approximately $\pm 10^{\circ}$, in a plane approximately perpendicular to the orbital plane.

Figure 3.3-18 also provides an example of how the previously described globographic presentation can be used to describe sun impingement angles with respect to the orbiter: Since the theoretical globe is considered to be firmly attached to the orbiter, a viewer on the orbiter sees the sun trace a different path along the surface of the theoretical globe on each orbit. The path is a function of the β angle of the orbit plane (angle relative to sun) and the attitude drift angle status of the orbiter. View A-A in Figure 3.3-18 shows how the average elevation of the sun's path (relative to the orbiter) may vary during the winter solstice (December 21). The orbital plane β-angle may vary from -52° to $+5^{\circ}$, depending on time of launch during the day.* The shaded area on the theoretical globe represents the entire range of possible sun positions in angular coordinates. Figure 3.3-19 indicates how such potential sun viewing angles can be plotted on the globographic presentation (shown as horizontal shaded area) and compared to the critical viewing angles for construction as shown in Figure 3.3-17. The resulting comparison in Figure 3.3-19 indicates that looking at the earth (starboard) and northward (overhead, away from payload bay) are favorable directions to avoid a direct view of the sun for conditions stated and the project being constructed. The orientation also satisfies one of the guidelines for orientation; that is, the payload bay generally faces the northern horizon, so that considerable reflected light from the earth

*See Figure 3.3-18



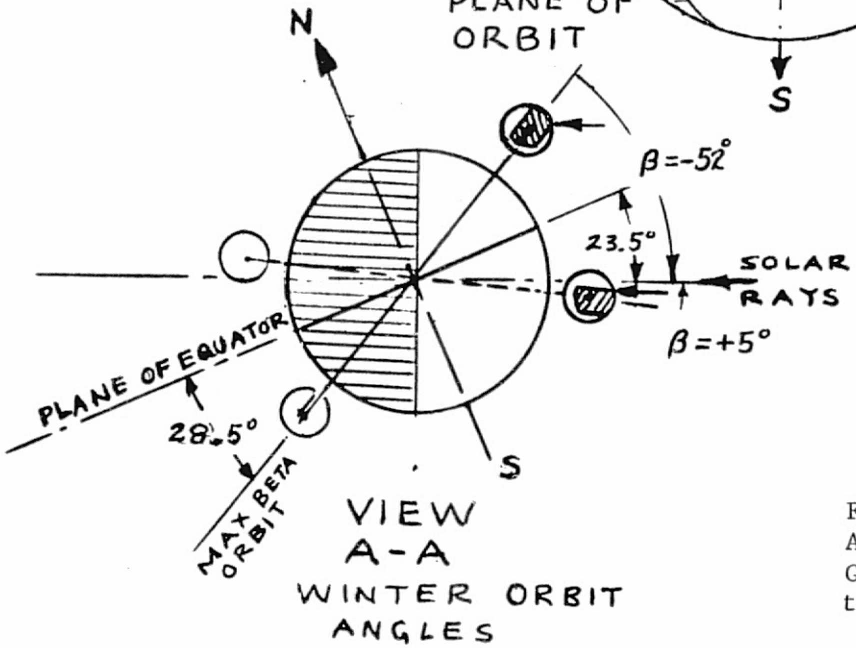
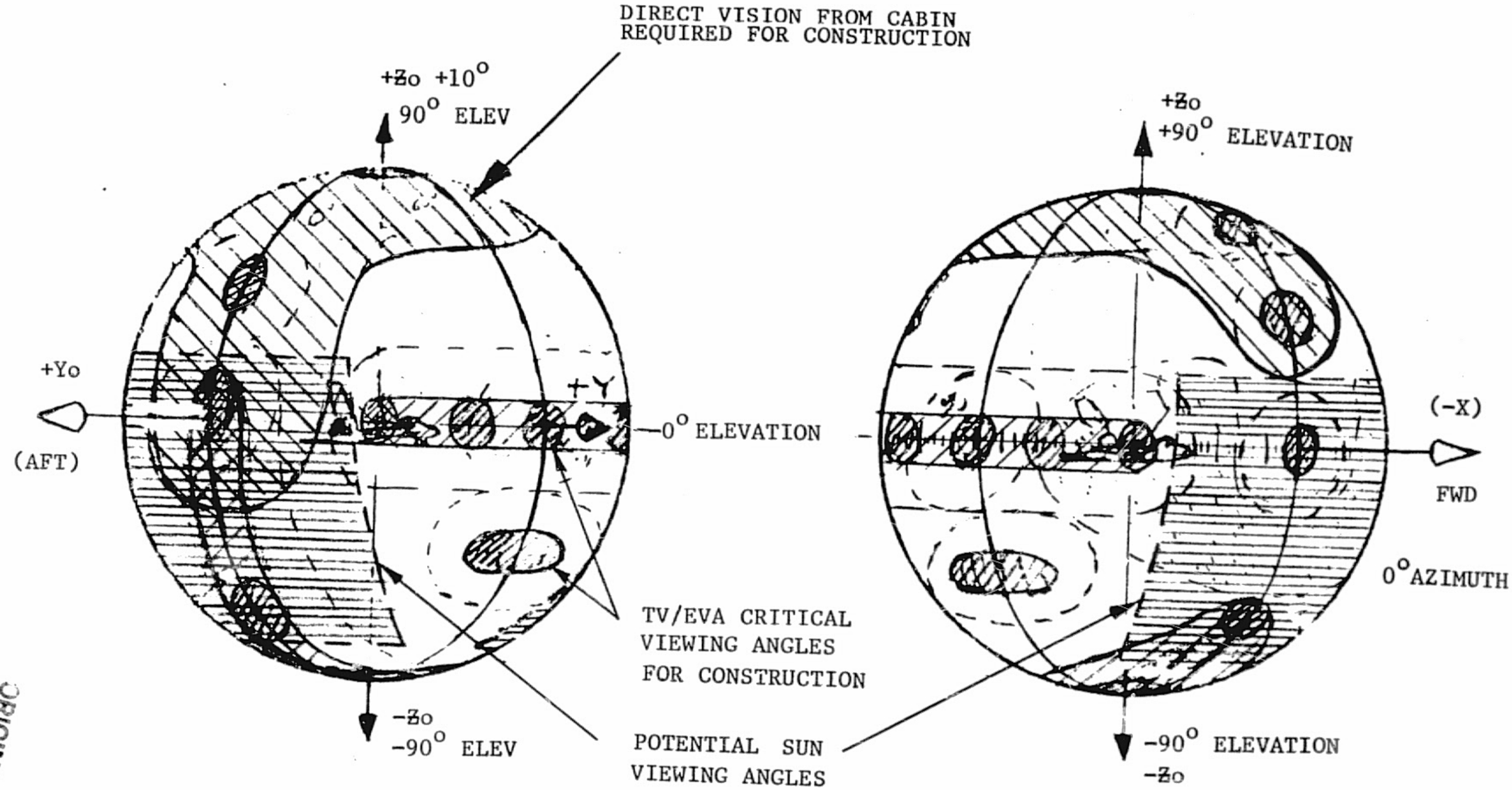


Figure 3.3-18 Example of Construction/Orbiter Attitude Drift Range and Relationship of Theoretical Globe for Describing Vision & Sun Angles Relative to Orbiter.



REAR QUARTER VIEW

FORWARD QUARTER VIEW

Figure 3.3-19 Solar Viewing Angles Compared to Key Viewing Angles for Construction illuminates at least one side of the structure mounted "above" the payload bay. For those orbits having β angles of approximately 20^{o} to 52^{o}, the orbiter structure provides considerable shielding against viewing the sun, since the belly is oriented somewhat toward the sun. The globographic diagram can also be used to illustrate the favorable range of angles to shield a specific viewing site from the sun. For example, Figure 3.3-20 compares the benefits of orbiter structure shielding to the possible range of solar viewing angles previously described. In the example, two different work sites are assumed. One site is the orbiter aft cabin, where considerable head movement is assumed possible so as to increase the field ofview out the windows beyond that usually shown. The visually obstructed area is shaded with vertical lines. The other viewing site is a fixed point about 20 meters above the center of the payload bay. The resulting visual obstruction area is shown shaded with horizontal lines. Figure 3.3-20 also shows the potential range of sun viewing angles outlined by a dashed line. A relatively small portion of this sun-viewing angle range is not shielded from the crew in the cabin (the area shaded by dots). However, the view from the site above the payload bay has a much larger chance of sunlight viewing. If this site represents a TV camera which is primarily oriented toward the payload bay and is shaded to prevent an excessively wide range of viewing angle, then it could be largely protected from sun viewing by the orbiter body.

This example of viewing angle analysis suggests that relatively favorable natural illumination conditions could be achieved by scheduling and taking advantage of construction project orientations resulting from gravity gradient stabilization alone. The prime considerations involve freedom to select the season and the time of launch for a particular construction session. One should select launch times which place the edge of the orbit facing the sun as far northward as is reasonably feasible. Winter operations would aid this trend. If the tail is leading, the sun is largely obscured by the wings and body at dawn and during the first half of the sunlit side of orbit, and the sun is behind the operator during the second half of the sunlit period. At the same time, during a large portion of the sunlit period, there will be diffuse earthshine on one or the other side of the construction surfaces. However, optimizing the intensity and variety of directions of such earthshine would tend to lower the edge of the orbit facing the sun toward a beta angle of zero degrees (plane of orbit parallel to sun's rays).

If construction operations must be scheduled for the summer season, it appears that it would be better to orient the orbiter with the nose pointing in the direction of travel and to select launch times which place the edge of orbit toward the sun as far south as feasible.

Thermal radiation from the orbiter radiators will be generally good in the attitudes described above, since there is little or no sun impingement on them and their orientation is primarily facing north or south, tangent to the earth's surface.

Rational Rate Considerations in Natural Illumination

Some concern may arise relating to a constantly changing visual environment during space construction. Possible problems are simple annoyance, inconvenience, confusion of shapes or directions, disorientation and even motion sickness.

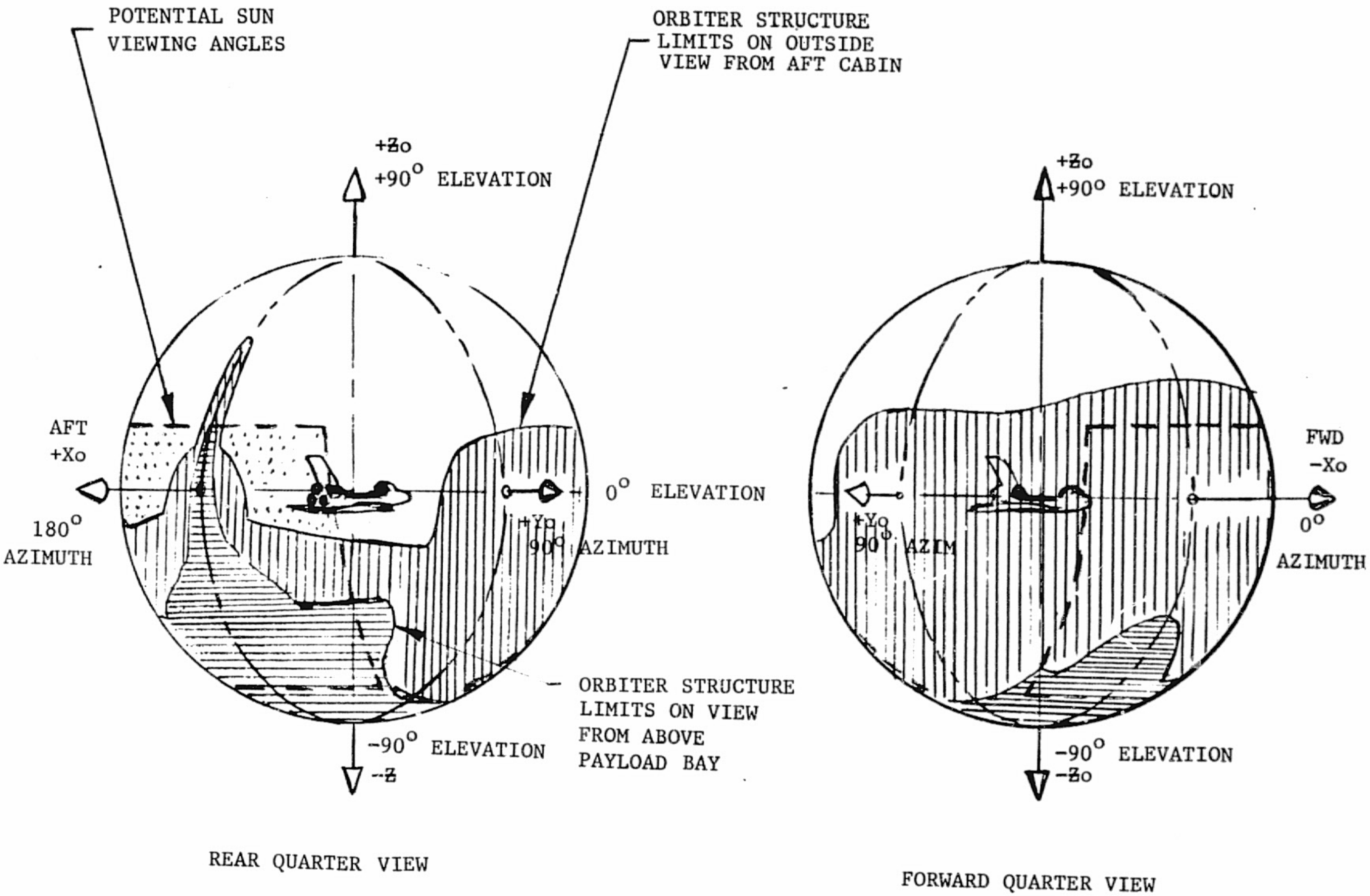


Figure 3.3-20 Solar Viewing Angles Compared to Orbiter Shading Potential From Two Viewing Sites

Most of these problems relate to the rates or motion experienced. In the specific example studied, the rotational rates were relatively slow, as shown in Table 3.3-23. Experience shows that such rates would generally have negligible effects on the crew as concerns disorientation or motion sickness. Larger rates could be experienced if the construction proceeds at a faster rate, or other balance conditions produce instabilities. The rates shown in Table 3.3-23 are not to be considered either average nor limiting conditions. However, even if the rates were ten times those listed, there is little chance of visual disorientation for the crew.

Table 3.3-23

Rotational Rates During Space Construction of Tri-Beam, Space Fabricated Communications Platform

(Angles Relative to Sun -- One Example Case)

Motion: Satellite Axes	Motion: Orbiter Axes	Rates (Degrees/Sec)
Roll	Roll	-.07 +.05
Pitch	Yaw	+.06 -.06
Yaw	Pitch	+.075 -.070

Finish Coating: Colors and Specularity

Non-specular (flat) finish coatings are highly desirable for viewing on the sunlit orbit as well as on the eclipse side. However, there is strong reason for somewhat darker (non-white) hues with lower reflectances on the sunlit side than on the dark side. This type of finish would give greater assurance that structure could be seen against bright white clouds or the white wing or payload bay surfaces on the orbiter. C. Wheelwright of NASA, Johnson Space Center, recommends colors in the yellow, yellow-green or brown range, with the yellow-green color of zinc oxide primer considered as the ideal. Rather than widely different hues, he recommends varied purity and chroma for contrast in differing structures and modules.

The currently planned TV camera for the Space Shuttle Orbiter RMS Payload Bay is a black and white type. Possible future developments could include color for these cameras for use in sunlight and would be highly desirable for viewing complex structural trussworks with color coded surfaces. Color coding could then be used to aid assembly and installation activities. However, it is recognized that such color coding would not be compatible with use of current night vision goggles or lowlight level TV cameras for use on the eclipse side of orbit. Therefore, if such devices are to be used, it seems unlikely to be cost effective to actively pursue development of color coding concepts and color TV as a general case. Rather, various high contrast markings concepts should be applied, such as lines, dots, stippling and texture variations, for key areas which must be differentiated visually.

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3.4 ELECTRICAL POWER

Electrical power and energy are among the most important support services. They are particularly important for construction concepts designed for implementation out of the orbiter. Electrical power is used directly and indirectly in almost every facet of space construction. It is used for illumination and crew support as well as to power various types of construction equipment and aids. These characteristics make it more of a construction integration issue than a construction concept problem. Thus, power considerations will be rigorously addressed during part two of the study. Here, emphasis is on the major significance of important power levels and the high-power users.

Table 3.4-1 presents preliminary estimates of the peak power requirements for various elements of the overall construction process. They include current projections for the beam machine, remote manipulator system (RMS) and the manned remote work station (cherry picker). The construction fixture and illumination estimates represent a mixture of extrapolation and buildups from preliminary data. Construction command and control and construction checkout tend to be dominated by the integrated construction process and cannot be readily defined for these early analyses. Although they are preliminary in nature the peak power levels listed here are sufficient to permit the identification of power and energy issues and their implications on the overall construction process.

Table 3.4-1. Construction Power User Summary

	POWER REQ'D (MAX)	
	SPACE FAB	ERECTABLE
BEAM MACHINE	2 KW	N/A
CONST. FIXTURE WELDERS TRANSLATION SWING ARM	2-5 KW	2-3 KW
RMS	1.8 KW	1.8 KW
MANNED REMOTE WORK STATION	0.5 KW	0.5 KW
CONST. COMMAND & CONTROL	TBD	TBD
ILLUMINATION	2-3 KW	3 + KW
CONST. CHECKOUT	TBD	TBD

Nearly all of these power users tend to be intermittent, ranging from either an "off" or "standby" condition to an operate condition. Examples of this are the beam machine and the RMS. Neither are used continuously throughout the mission. How the intermittent loads dovetail is the main issue. In reviewing the possible combinations of usage the following observations can be made. When the beam machine is operated the construction fixture and the RMS would tend to be in low power modes. Similarly when the fixture is used for performing a structure translation operation, welding cannot occur. Also, construction checkout will tend to occur at times when the construction fixture is powered down. External lighting could be shut off during the daylight side of each orbit. This would reduce its power utilization to the 35 to 40 percent range depending upon sun/orbit geometry. Construction command and control will tend to be a continuous (but probably variable) function, but should not pose high power demands.

Thus, with prudent scheduling of the construction operations construction out of the orbiter appears feasible. The 7 kW continuous and 12 kW peak power available from the orbiter should satisfy the construction needs. The key issue is the power-energy relationship. If construction task scheduling for "peak power" management lengthens the mission beyond 7 days, the power required to operate the orbiter (approximately 13 kW) becomes payload chargeable. Depending upon the duration, the required extra cryo kits may exceed the space available beneath the cargo bay. This would occur for mission durations of 10 days or longer. Cryo kits for mission durations exceeding this value must be installed in the cargo bay thereby potentially affecting construction packaging. This issue will be fully investigated as part of the construction integration analysis in part two of the study.

Several differences are noted in Table 3.4-1 between the power requirements for space fabrication and erectable construction. These involve power for the construction fixture and illumination. The fixture concept for erectable construction is shown to require significantly less power than that for the space fabrication construction process. This is the result of basic process differences. The main functions of the erectable fixture are retention and translation of structural elements. The space fabrication fixture also has these functions, but must also provide for positioning and welding of fabricated elements. Thus, its power demands will be higher.

Power requirements for illumination could be greater for the erectable case. The basic platform structure for the erectable design is based on 12 meter struts compared to the approximately 7 meter envelope for the space fabricated tri-beam platform. Thus, the larger construction area associated with the bigger structural envelope of the erectable concept could require more power for illumination.

The combined effects of these two factors, fixture power and power for illumination, depend upon the integrated construction processes for each concept and cannot be finalized at this point in the study. However, the significant differences in fixture power coupled with no requirement for beam machine power suggests that erectable construction concepts may require less power than space fabrication approaches. These are preliminary results and more analyses are necessary. Integrated power requirements will be fully analyzed in part two of the study.

4.0 CONSTRUCTION FACILITY IMPLICATIONS

4.1 INTRODUCTION

This discussion addresses the construction of a large area platform type of project from a permanently manned facility in low earth orbit (LEO). The LEO facility model used in this discussion is a Space Operations Center (S.O.C.) concept developed by Jim Jones of NASA/JSC. This concept is illustrated in Figure 4.1-1.

The purpose of this document is to discuss the S.O.C.'s construction facility potential capabilities, to indicate the implications on the facility, and to list guidelines that will permit the S.O.C. to fulfill its potential capabilities. Considerations addressed include those associated with the construction effort such as the construction support equipment, services, logistics and crew issues; the effect of space construction from the S.O.C. on the Shuttle Orbiter, and the effect on the construction project.

For purposes of this discussion the S.O.C. is assumed to be located in a 250 nmi circular orbit at an inclination of 28.5 degrees. The Shuttle Orbiter delivers construction support equipment, project components and general logistics supplies for operation of the S.O.C.. In addition to being a construction facility, the S.O.C. provides facilities for science and applications experiments and for orbit transfer vehicle (O.T.V.) staging/refueling.

The facility is permanently manned with the capability to accept a construction work crew for the fabrication and assembly period required for a particular construction project.

4.2 FUNDAMENTAL POTENTIALS OF SPACE CONSTRUCTION FROM THE S.O.C.

There are three principal subjects to be considered when discussing construction from a S.O.C.: (1) the orbiter logistics operations, (2) the construction operations, and (3) the construction project design.

4.2.1 Orbiter Logistics Operations

The orbiter operations issue includes the logistics associated with the construction operations and the overall Shuttle Orbiter operations considerations.

With all of the construction fixtures and special construction devices located and operated from the S.O.C., the orbiter becomes only a means for transporting supplies to the facility. A baseline orbiter can fulfill this requirement without the necessity for any operational modification kits, with the possible exception of an Orbit Maneuvering System (OMS) kit if

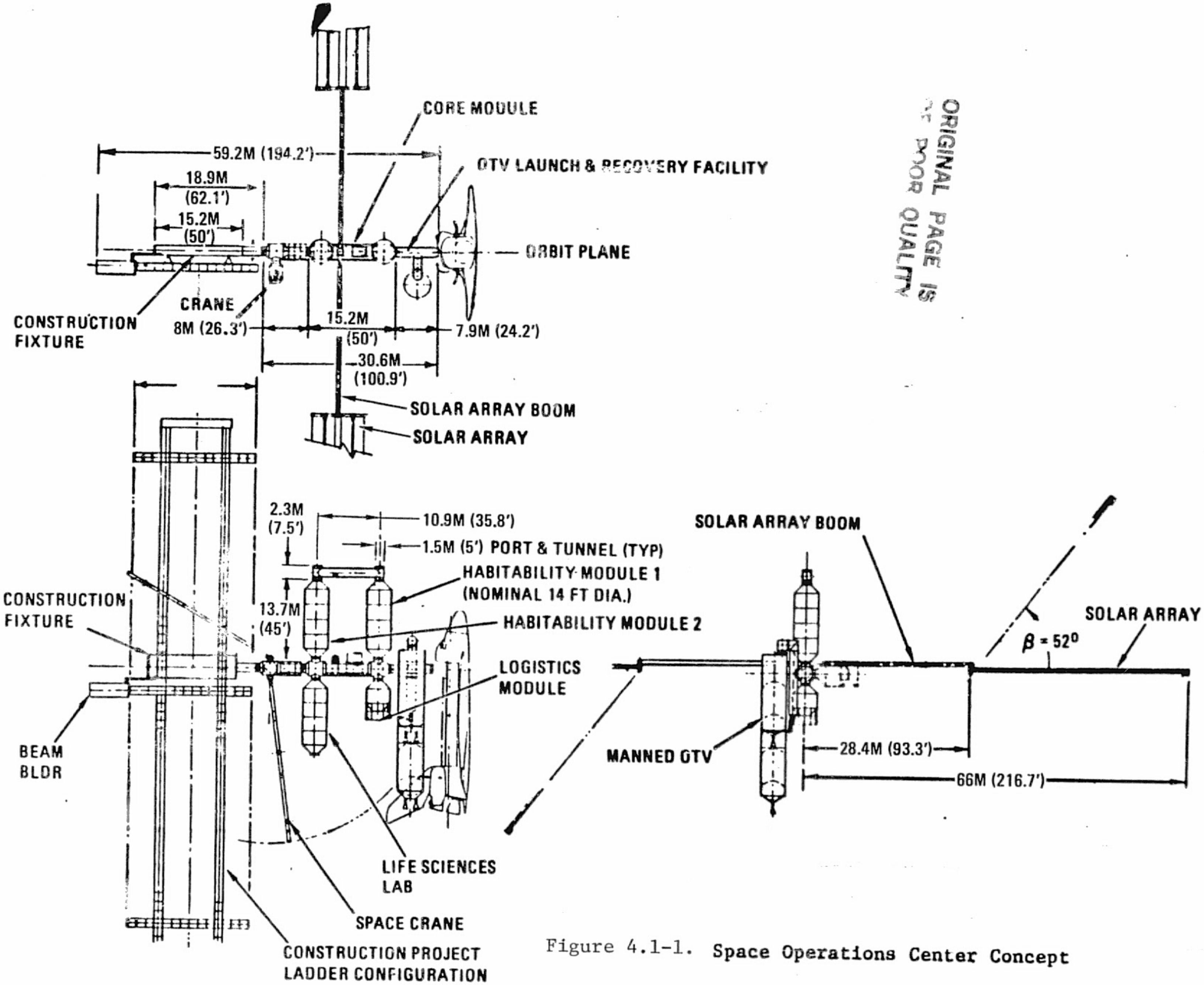


Figure 4.1-1. Space Operations Center Concept

required as a result of the operating altitude of the S.O.C. Consequently, a dedicated orbiter is not required for construction operations.

This condition, therefore, allows flexibility in scheduling Shuttle operations and improves the utilization of the Shuttle fleet and the ground support operations.

Because the Shuttle Orbiter provides only a cargo delivery mode, the orbiter crew does not require any specialized training associated with space construction activities. Space construction operations training can be reserved exclusively for the S.O.C. construction crew.

In summary, the orbiter and the delivery/ground operations logistics can be optimized because the orbiter provides a cargo delivery mode only and, therefore, does not require a dedicated orbiter.

4.2.2 Construction Operations

Construction operations from the S.O.C. has the potential for continuous fabrication and assembly. Simultaneous/parallel construction operations can be accommodated in the S.O.C. as contrasted to limited serial operations when performing construction from the orbiter. The continuous and parallel operations potential will result in more productive utilization of the facility and, consequently, minimum construction time and early operational status of the construction project.

Construction from the S.O.C. appears to have the potential for more productive crew activity because (1) the crew has been specifically trained for this operation, (2) will have experience in space construction and (3) have a minimum of interruptions in the construction process. (Construction operations from the orbiter are interrupted by "start-up and shut-down" activities.) Increased proficiency can also be assumed due to the more comfortable crew accommodations afforded by the S.O.C.

The permanent construction facility as defined by the S.O.C. permits the use of more mechanized and automated construction methods. This is possible because of the available space with better visibility and the manipulator or space crane that has long reach capability. The construction fixture need not be designed to be folded for easy erection from the orbiter, but can be assembled at the S.O.C. with individual elements. This capability also allows for greater mechanization to be implemented. Electrical power for these operations is also available.

The S.O.C. also has the potential for accommodating multiple beam builders and other universal types of construction support equipment. The beam builders can be assumed to be at the S.O.C. and available for construction operations. The availability of multiple beam builders increases the productivity and may simplify the construction/assembly process. With the beam builders on site, the orbiter need only deliver the material canisters required by the beam builders. This potential operational concept also contributes to a better payload bay packaging efficiency because the beam builders are not transported to and from the S.O.C. for each construction project.

The potential for an on-board system checkout facility may be available. With this capability available the transport of special units to accomplish this activity would not be necessary.

4.2.3 Construction Project Design

With the potential for greater mechanization of the construction process, less reach restrictions with the space crane, and overall space availability, design constraints of the construction project can be minimized. The design constraints referred to are those that are associated with the available equipment planned for construction support such as the orbiter RMS. The reach envelope of the RMS influences the design of the overall dimensions of the construction project fabricated and assembled from the orbiter. The RMS capability also influences systems installations concepts. Figure 4.2-1 illustrates the space fab tri-beam configuration of the advanced communications platform that is an example of the RMS reach constraints. The tri-beam size is limited by the requirement that the RMS must be able to reach the ends of the cross beam where antennas and RCS modules are installed. With the availability of a space crane fabrication and assembly of a construction project at the S.O.C. significantly minimizes these design constraints.

The overall size of the construction project need not be constrained because of the construction fixture size that can be erected from the orbiter with minimum complexities. Figure 4.2-2 illustrates a revision to the project configuration of the erectable advanced communications platform assembled from the orbiter in order to permit project translation with minimum construction fixture complications. RCS moment arm, however, was compromised as a result of the revision. The potential exists for larger fixtures to be assembled at the S.O.C., thus minimizing these types of project constraints.

In summary, the construction project has less design constraints associated with the construction facility because of the S.O.C.'s potential capabilities.

Table 4.2-1 summarizes the fundamental potentials of construction from the S.O.C.

4.3 IMPLICATIONS AND SYSTEM GUIDELINES

The previous section discussed the potential capabilities of the S.O.C. to accomplish space construction. This section addresses the design requirements necessary to implement the capabilities.

4.3.1 S.O.C. Logistics

With the orbiter delivering cargo to the S.O.C. for the construction operations, the S.O.C. must provide the capabilities to accept and store the cargo. In order to minimize the orbiter stay time the cargo should be transferred as rapidly as possible. This requirement suggests that the cargo be packaged on "pallet" type supports that can be lifted from the orbiter payload

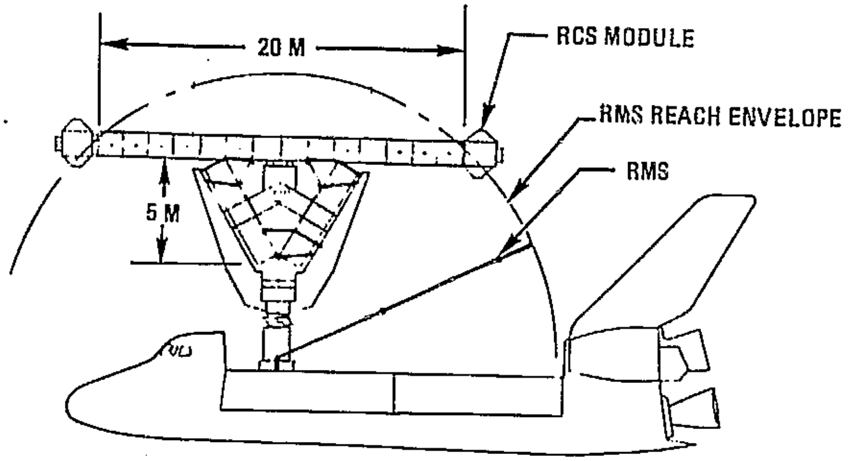


Figure 4.2-1. RMS Reach Constraints on Project Configuration

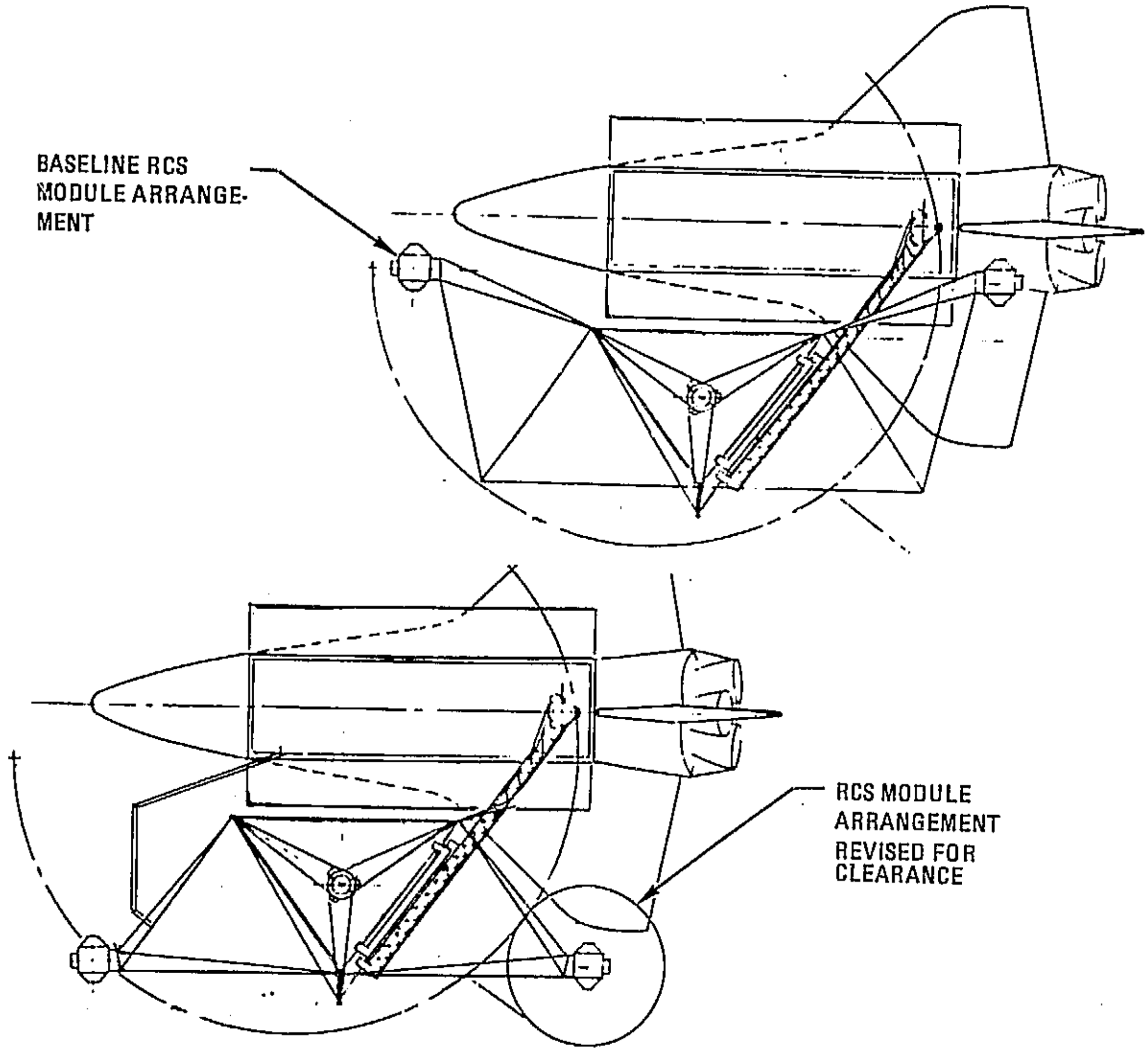


Figure 4.2-2. Construction Fixture Influences on Project Configuration

Table 4.2-1. Summary - Fundamental Potentials of Space Construction from the S.O.C.

ORBITER LOGISTICS OPERATIONS

No modifications required for Orbiter
Improved utilization of Shuttle fleet
Potential for reduced logistics cost

CONSTRUCTION OPERATIONS

Shortened construction time - earlier return on investment
Improved productivity per space man-hour
Improved mechanization of construction fixtures
Inventory carry-over to other construction projects

CONSTRUCTION PROJECT

Reduced construction constraints - improved project design
Larger scale projects may be accommodated

bay to the S.O.C.. Two areas, as a minimum, must be provided on the S.O.C. to accept these cargo pallets. One area accepts a full pallet and the other area stores the empty pallet making it available for return.

The pallet storage areas must be near the construction site in order to reach the materials and components for assembly. A separate storage area is also required near the construction site for the storage of the beam builders when they are not being used and other standard construction support equipment (e.g., manned remote work station, (MRWS), manned maneuvering unit (MMU), etc.,) and standard construction tools.

4.3.2 Construction Operations

In order to utilize the potential for continuous construction certain requirements are suggested. The material and construction equipment must be available in order to implement the continuous construction operation. The two storage areas described above would be utilized in order to facilitate this requirement. Since construction is only one of the functions that will be performed by the S.O.C., any required additional storage area could be shared with other functions.

Docking the orbiter in the vicinity of the construction fixture for the delivery of pallets containing construction materials and components is desirable. By locating adjacent to the construction operations area, the orbiter RMS is potentially capable of performing the unloading task, thus freeing the space crane for construction operations without any interruptions from the orbiter delivery operation.

The continuous construction operations also requires more crew members for an around-the-clock operation, which would suggest a three shift operation. Crew accommodations, therefore, will be required to house, feed, and maintain this crew during construction. A three shift operation also requires additional design considerations for the isolation of activity and quiet areas. Sleep areas isolated from eating, exercise, and operations activity areas are desirable in order to maintain proficient crew operations.

A construction operations control facility should contain the controls for the operation of the construction devices, control the illumination, and provide checkout equipment. A master control facility that would contain these functions as well as the space crane control and the shirt-sleeve environment control, airlock, MMU servicing, etc., would also be a desirable feature. This control facility would also provide the storage and servicing of the EVA suits. Communications throughout the construction operations would be controlled from the construction operations control facility with the possibility of ground communications being routed via the S.O.C. communications system. Good observation of the construction operations both direct and via T.V. from the construction operations facility is necessary.

Transport of the construction projects to their operating altitude will require an OTV staging and assembly area facility at the S.O.C.. This area would contain berthing facilities for the propulsion modules in such a location as to permit the space crane to install them on the construction project.

During the construction operation, the S.O.C. may be resupplied with RCS propellant to maintain the attitude control and orbit altitude. Therefore, appropriate storage and refueling capacity must be available.

4.3.3 Construction Fixture Facility

In order to take full advantage of the potential construction fixture flexibility afforded by the S.O.C., a universal type platform to which the fixtures/devices can be mounted would appear to be a desirable feature. The universal mounting platform could provide the basic fixture structure. Individual supporting devices, assembly devices, etc., could be mounted to the platform. This concept could reduce the number of Shuttle flights because the individual devices would package more efficiently and the basic mounting structure being on the S.O.C. would not require transport.

Table 4.3-1 summarizes the implications and guidlines for the S.O.C. to accommodate space construction operations.

4.4 CONCLUSION

Space construction from a S.O.C. has the potential advantage of simplifying the orbiter's role to only that of a cargo transport vehicle. This permits the standard orbiter to perform this function and no individual dedicated orbiter is required for space construction. Cargo secured to "pallets" appears to be a desirable concept for this operation because it permits rapid unloading and turn-around operations, thus minimizing stay-time and releasing the orbiter for other missions.

The S.O.C. requires facilities to accept the cargo pallets in the vicinity of the construction operations. At least two areas for the retention of the pallets are required in order to deliver a full pallet and return an empty pallet. Storage facilities are also required for the storage of beam builders and other universal support equipment which can be retained at the S.O.C.

The potential capability for continuous construction activities is desirable to minimize construction time, and provide more proficient operations. Crew accommodations will be required for a larger crew when implementing this capability.

Specialized crew training limited to construction operations, in conjunction with more comfortable accommodations at the S.O.C., creates the potential of developing a more proficient crew for this operation.

The potential capability to have more mechanized/automated construction methods with the S.O.C. is available with the inherent clear area available for construction, better direct visibility and the availability of the space crane. The available electrical power also makes the greater mechanization feasible.

Table 4.3-1. Summary - Implications & System Guidelines for Construction Operations

LOGISTICS:

- Provide storage facilities near construction site for materials, components, etc., and for standard construction support equipment
- Transport cargo in removable pallets
- Provide docking facilities for orbiter near construction site

CONSTRUCTION OPERATIONS:

- Provide an OTV staging/assembly area near construction site
- Provide a space crane
- Provide crew accommodations
- Provide a construction operations control facility for
    - Control of all construction functions
    - EVA suit storage and servicing
    - Communications control
    - Good observation of construction activities
- Provide attitude control capability
- Provide an airlock near construction site

CONSTRUCTION FIXTURE FACILITY

- Provide a facility to mount construction fixtures and various construction devices

Construction from the S.O.C. minimizes construction project design constraints because of the greater reach envelopes of the space crane, and the clear area available for the construction operations.

In summary, space construction operations from a S.O.C. facility have potential significant advantages over space construction from the orbiter as a construction facility.

5.0 ORBIT TRANSFER ANALYSIS

This section presents the results of two topics of vital importance to space construction, but whose influences are somewhat indirect to the actual construction process. The first topic is construction orbit altitude (Section 5.1). Drag/orbit decay, EVA radiation hazard and orbiter logistics performance interact to determine safe construction orbit altitudes for each type of project system. Orbiter logistics performance can have a major effect on packaging requirements and, hence, can affect the construction sequence.

The second topic is orbit transfer (Section 5.2). Various propulsion concepts were analyzed to determine their suitability for LEO-to-GEO orbit transfer of large area space systems. Impacts of thrust loads, control frequency/structural stiffness interactions, and basic propulsion stage performance and sizing were considered. Alternative techniques for delivering propulsion modules to the construction orbit were also analyzed.

5.1 CONSTRUCTION ORBIT ALTITUDE

There are a number of significant factors which can affect the orbit altitude selected for space construction. These include drag/orbit decay, radiation environment/EVA, and orbiter delivery performance in terms of both weight and volume. All of these factors have a strong dependence on the project/configuration to be constructed. The drag characteristics are dependent on the shape and area features of the particular project as is the construction duration, both of which contribute to the amount of orbit decay likely to occur during the construction process. The particular project configuration will also affect the attainable packaging factor in the orbiter bay loading for each of the construction missions. This, combined with Shuttle performance limits, restricts the construction orbit altitude to that associated with the "heaviest" bay loading condition for each particular project, or may introduce the need for OMS kits which can impact bay packaging and the number of flights.

Not all of these factors can be rigorously treated here for all possible projects and configurations. However, example results treating the Advanced Communications and SPS Test projects are presented to establish the overall importance of construction orbit altitude, and to illustrate the relative significance of these main parameters and how they are interrelated. There is no "optimum" orbit for all space construction activities. Each project will have its own preferred construction orbit with specific margins and mission operations suitable to its particular needs.

The following paragraphs, then, present the primary findings of this study.

5.1.1 Orbit Decay

One of the main concerns in selecting the orbit altitude for space construction is orbit decay. The new large area space systems made possible by the Space Shuttle are designed to be constructed in space and need only to be sized for very small forces and loads, and are typically very light in weight. Hence, they tend to have very low ballistic coefficients (W/C_DA) which can result in relatively high rates of orbit decay. To illustrate the severity of this problem, several example situations were investigated for the project systems defined in this study. A range of ballistic coefficients were calculated for each project, representing its different area/weight features through the basic construction process. These were combined to form a "drag" profile representing the overall construction process, including periods with and without the orbiter. The maximum and minimum ballistic coefficients were averaged to determine orbit decay characteristics associated with construction in a "free drift" mode in which the orientation would tend to be random. Actually, the orientation history would have gravity-gradient tendencies to oscillate about changing principal axes, as discussed in Section 3.1. However, for this analysis it was presumed that the orientation excursions and rates associated with this dynamic motion would be sufficiently large that all possible "drag" orientations would occur and hence could be approximated by averaging the frontal area about all three body axes.

SPS Test Article Decay Profile

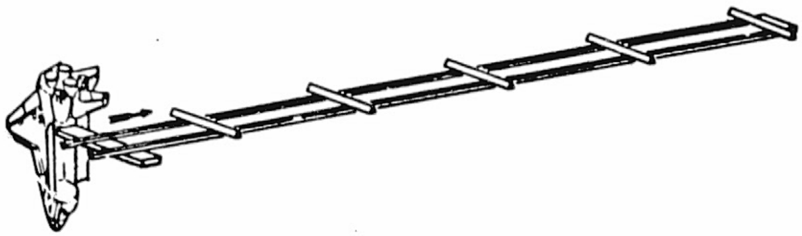
A sequence of configurations representing the "build up" of the operational project system was assumed for this analysis. The actual construction sequence will be dependent upon future trades and evaluations of alternative construction methods and processes, and could exhibit detailed differences from those assumed here. However, for purposes of developing an initial understanding of orbit decay effects on construction orbit altitude, this preliminary configuration sequence is believed to be adequately representative.

The assumed sequence is as follows:

- Step (1) Construct all structures and lay in electrical power distribution system, lines, J-boxes, etc.
- Step (2) Install RCS modules
- Step (3) Install solar blankets
- Step (4) Install subsystems module and the microwave antenna

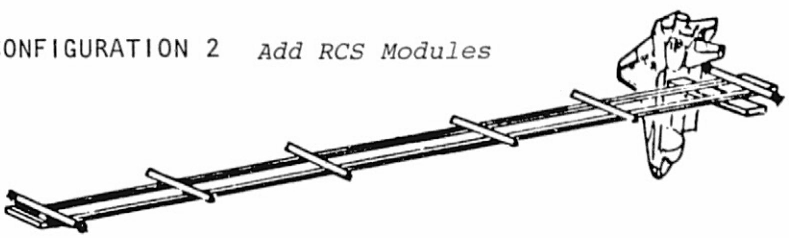
These construction steps are shown pictorially in Figures 5.1-1 and 5.1-2. The operational configuration is shown in Figure 5.1-3. Weights are listed along with the range of ballistic coefficients for each configuration. Minimum, maximum, and average values are presented. Average values for the ballistic coefficients are based on a simple sine wave averaging concept. In this concept the frontal areas for the major drag surfaces of each configuration are averaged with the following expression: $A_{avg} = A_{min} + 0.6366 (A_{max} - A_{min})$, where the constant 0.6366 is the average height of a sine wave

CONFIGURATION 1 *Structure plus Electrical Lines*



	WITH ORBITER	WITHOUT ORBITER
WEIGHT	89350 KG (197,000 LB)	16,782 KG (37,000 LB)
$(W/C_DA)_{min}$	96.1 KG/M^2 (19.7 PSF)	63.9 KG/M^2 (13.1 PSF)
$(W/C_DA)_{max}$	298.1 KG/M^2 (61.1 PSF)	79.5 KG/M^2 (16.3 PSF)
$(W/C_DA)_{avg}$	158.1 KG/M^2 (32.4 PSF)	68.8 KG/M^2 (14.1 PSF)

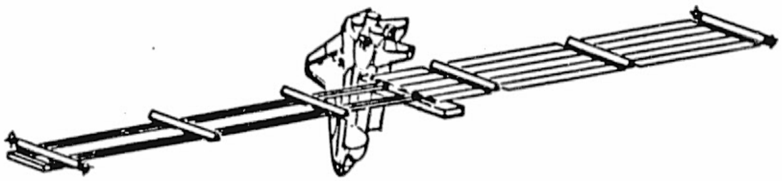
CONFIGURATION 2 *Add RCS Modules*



	WITH ORBITER	WITHOUT ORBITER
WEIGHT	94,793 KG (209,000 LB)	22,224 KG (49,000 LB)
$(W/C_DA)_{min}$	102 KG/M^2 (20.9 PSF)	84.4 KG/M^2 (17.3 PSF)
$(W/C_DA)_{max}$	316.2 KG/M^2 (64.8 PSF)	105.4 KG/M^2 (21.6 PSF)
$(W/C_DA)_{avg}$	167.9 KG/M^2 (34.4 PSF)	91.2 KG/M^2 (18.7 PSF)

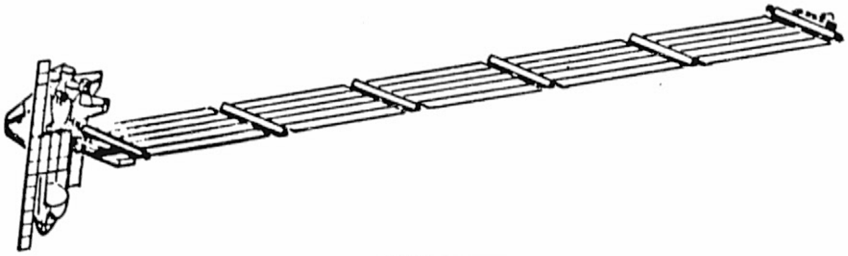
Figure 5.1-1. SPS Construction Configurations 1 and 2

CONFIGURATION 3 *Add Solar Blankets*



	WITH ORBITER	WITHOUT ORBITER
WEIGHT	100,236 KG (221,000 LB)	27,667 KG (61,000 LB)
$(W/C_DA)_{min}$	12.0 KG/M^2 (2.46 PSF)	3.3 KG/M^2 (0.68 PSF)
$(W/C_DA)_{max}$	38.5 KG/M^2 (78.9 PSF)	106.4 KG/M^2 (21.8 PSF)
$(W/C_DA)_{avg}$	17.6 KG/M^2 (3.60 PSF)	5.1 KG/M^2 (1.05 PSF)

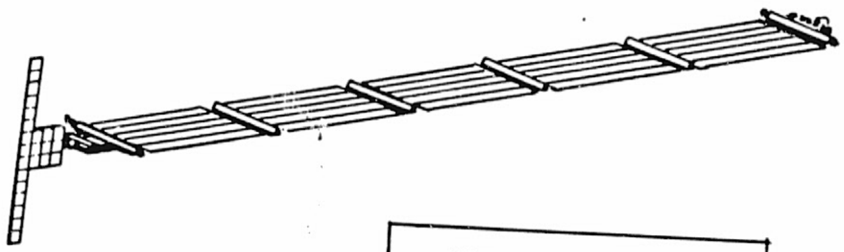
CONFIGURATION 4 *Add SS Module and Microwave Antenna*



	WITH ORBITER	WITHOUT ORBITER
WEIGHT	560,019 KG (254,000 LB)	42,634 KG (94,000 LB)
$(W/C_DA)_{min}$	13.2 KG/M^2 (2.7 PSF)	5.1 KG/M^2 (1.04 PSF)
$(W/C_DA)_{max}$	108.3 KG/M^2 (22.2 PSF)	127.4 KG/M^2 (26.1 PSF)
$(W/C_DA)_{avg}$	20.1 KG/M^2 (4.12 PSF)	7.6 KG/M^2 (1.56 PSF)

Figure 5.1-2. SPS Construction Configurations 3 and 4

CONFIGURATION 5 *Operational Configuration*



	WITHOUT ORBITER
WEIGHT	37,781 KG (83,300 LB)
$(W/C_DA)_{min}$	4.4 KG/M^2 (0.9 PSF)
$(W/C_DA)_{max}$	156.1 KG/M^2 (32.0 PSF)
$(W/C_DA)_{avg}$	7.0 KG/M^2 (1.44 PSF)

Figure 5.1-3. SPS Test Article Operational Configuration

with an amplitude of 1.0. Although this expression would be exact only for a flat plate surface, it was felt to be sufficiently accurate for the preliminary analyses here that it was applied to the three-dimensional shapes herein. The basic data used in the ballistic coefficient calculations are summarized in Table 5.1-1.

Several orbit decay profiles were generated based on this configuration sequence and its drag characteristics. These profiles were based on the decay rates shown in Figure 5.1-4. This shows the decay time (days) required for a 10-percent drop in orbit altitude as a function of altitude and ballistic coefficient. These data are for a nominal solar maximum atmosphere as shown on the insert in Figure 5.1-4.

The al tude drop for each step in the construction sequence was determined with these decay rate data. The resulting end altitude for each step was used as the initial altitude for the next step. Each step or segment of the overall construction sequence was assumed to be comprised of two basic parts. The first is a seven-day interval with the orbiter attached in which the actual construction operations are performed. The second part is a 23-day interval representing the coast time between Shuttle revisits where the drag configuration does not include the orbiter. Thus, an overall construction process involving Shuttle launches on 30-day centers was assumed. This is slightly conservative over the 14-day turnaround currently projected for Shuttle operations, but was felt to be appropriate for these preliminary analyses directed toward identifying the drivers affecting orbit altitudes for space construction.

The resulting decay profiles are shown graphically in Figure 5.1-5. Three initial altitudes were analyzed: 555 km (300 nmi), 509 km (275 nmi), and 463 km (250 nmi). In addition to the basic construction sequence described above, two additional factors were considered. First, a 30-day system checkout interval was added to the end of the decay profile to allow for verification of a properly completed construction process. The drag configuration for this segment of the decay profile was assumed to be that of the operational vehicle system (Figure 5.1-3), $W/C_DA = 7.03\ kg/m^2$ ($1.44\ lb_m/ft^2$). The second additional factor was the insertion of a 60-day "contingency" phase between the third and fourth step in the basic construction sequence. This could represent the occurrence of some type of problem within the construction process or within the orbiter turnaround operations. In a broad sense, it could also represent the cumulative effects of several smaller delays scattered along the construction sequence. These contingency extensions utilized the ballistic coefficients associated with the end of Step 3, which is $W/C_DA = 5.12\ kg/m^2$ ($1.05\ lb_m/ft^2$) and are shown as dashed lines in the decay profiles (Figure 5.1-3).

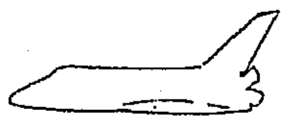
Nominal end construction altitudes are shown to range from 526 km (284 nmi) to 354 km (191 nmi) for initial orbit altitudes of 555 km (300 nmi) and 463 km (250 nmi), respectively. Extending the construction duration with the 60-day contingency period greatly affects the decay problem. For the 463-km (250-nmi) initial altitude case, the 60-day extension causes the orbit to decay completely. Thus, an initial altitude of 463 km is too low for space construction of the SPS test vehicle without some concept for orbit makeup and/or orientation control to minimize drag during construction operations.

Table 5.1-1. Basic Data for Ballistic Coefficients

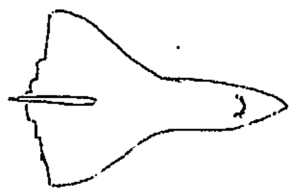
ORBITER WEIGHT = 72,570 KG (160,000 LB)

ORBITER DRAG AREAS:

$A = 41.8\ M^2$ ($450\ FT^2$)



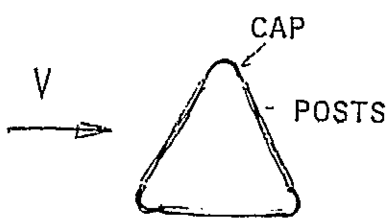
$A = 285.9\ M^2$ ($2000\ FT^2$)



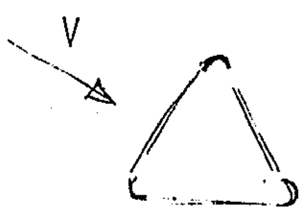
$A = 365.3\ M^2$ ($3930\ FT^2$)

SPACE FABRICATED BEAM DRAG AREA

AREA PER UNIT LENGTH



$A = 0.2\ M^2/M$ ($2.16\ FT^2/M$)



$A = 0.31\ M^2/M$ ($3.32\ FT^2/M$)

DRAG COEFFICIENT:

$C_D = 2.0$ BASED ON FRONTAL AREA

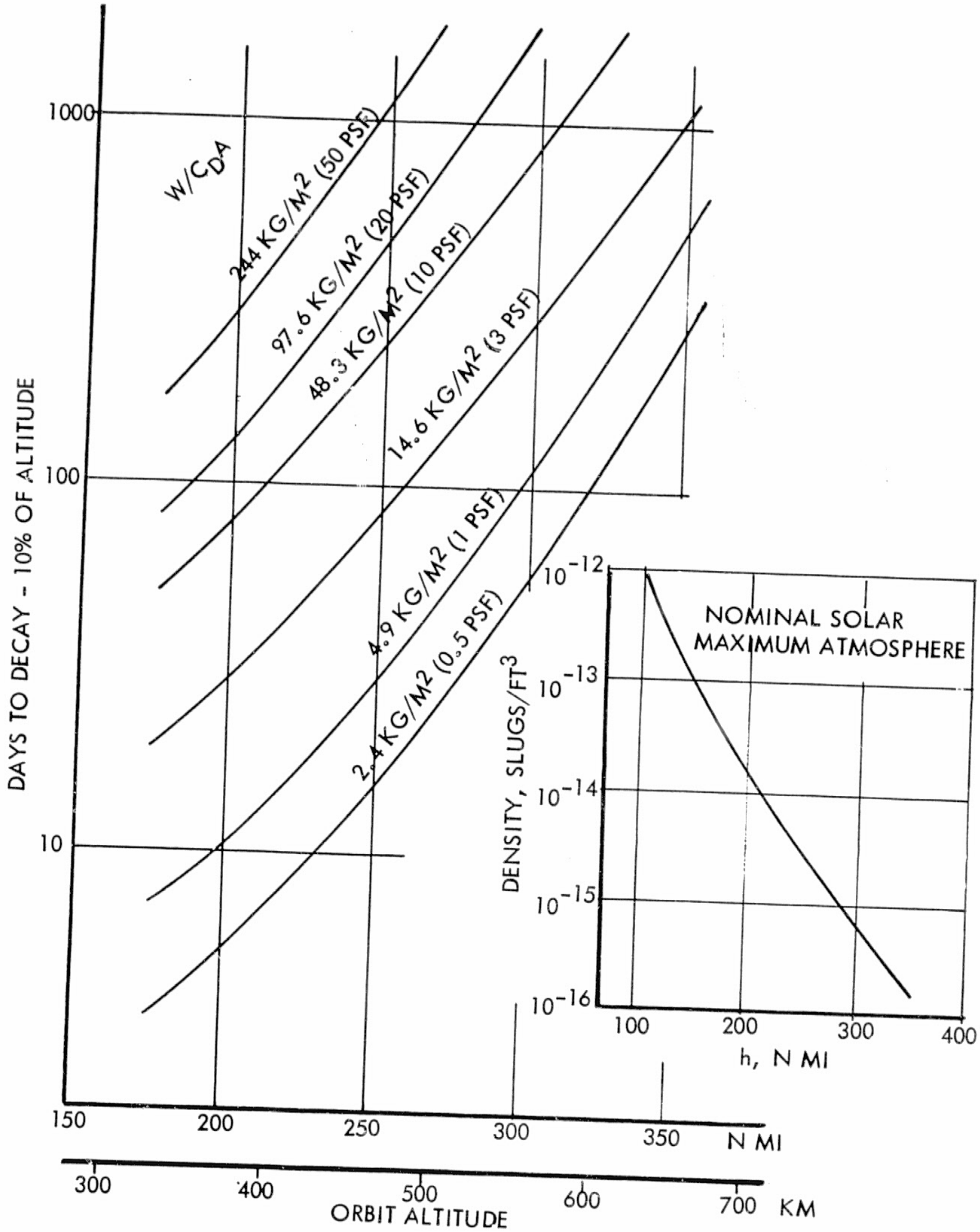


Figure 5.1-4. Parametric Orbit Decay Characteristics

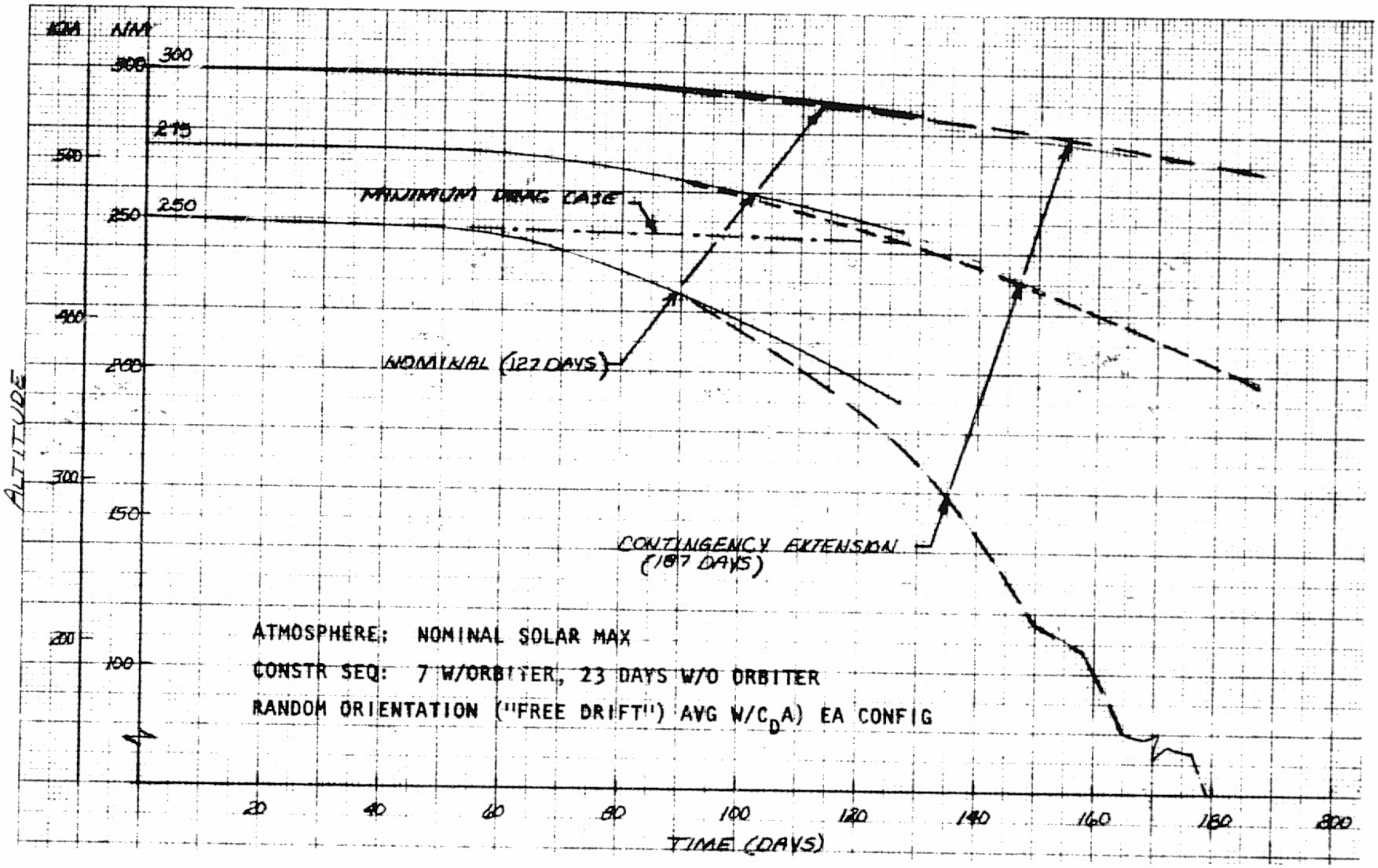


Figure 5.1-5. Orbit Decay During SPS Test Vehicle Construction

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To illustrate how powerful the effect of drag/orientation can be on the decay profile, a special minimum drag case was generated. In this case the orientation for each step of the construction was assumed to correspond to the minimum drag values (maximum ballistic coefficient) given in Figures 5.1-1 and 5.1-2. The resulting decay profile is shown in Figure 5.1-5 along with the three "average" drag decay profiles previously discussed. The decay rate for this case (initial altitude of 463 km) is only about 10 percent of the nominal profile and even with the 60-day contigency extension the final construction orbit altitude is 354 km (191 nmi). Thus, attitude control during construction can have very significant effects on the decay profile and, hence, on construction orbit altitude. These altitude/orientation benefits with attitude control must be weighed against the RCS propellant required to maintain attitude and possible complications in the construction process to accommodate RCS thruster induced loads and/or disturbances.

Advanced Communications Platform Decay Profile

To further explore the factors affecting construction orbit altitude, the decay profile for a second project system—the Advanced Communications Platform—was also investigated. This case serves to illustrate the differences that can exist due to individual project configurations.

The same basic incremental orbit decay process, based on a sequence of construction steps, was applied to the decay profile for this configuration as was used for the SPS test vehicle discussed previously. However, a new six-step construction sequence suited to the communications platform configuration was synthesized. This generalized sequence is summarized as follows:

- Step (1) Construct all structure and lay in electrical power distribution system, lines, J-boxes, etc.
- Step (2) Install rotary joint assembly and battery system module.
- Step (3) Install solar array canister assemblies and two RCS modules
- Step (4) Install eight antenna modules.
- Step (5) Install eight more antenna modules.
- Step (6) Install two RCS modules and the systems control module.

These construction steps are shown pictorially in Figures 5.1-6 through 5.1-8. Weights are listed along with the range of ballistic coefficients for each configuration.

In addition to these specific construction steps, two additional post-construction operational phases were considered. First, a 30-day checkout period was introduced in which the configuration was assumed to be that of the operational platform, top of Figure 5.1-9. The solar arrays and antenna systems would be deployed and the configuration would fly with the long axis perpendicular to the orbit plane. The average ballistic coefficient for this phase is: $W/C_DA = 22.5\ kg/m^2$ (4.62 psf).

CONFIGURATION 1 *Structure plus Electrical Lines*

	WITH ORBITER	WITHOUT ORBITER
WEIGHT	94,612 KG (208,000 LB)	22,043 KG (48,600 LB)
$(W/C_DA)_{min}$	73.7 KG/M² (15.1 PSF)	36.1 KG/M² (7.4 PSF)
(W/C_DA)	191.8 KG/M² (39.3 PSF)	52.2 KG/M² (10.7 PSF)
(W/C_DA)	94.7 KG/M² (19.4 PSF)	40.5 KG/M² (8.3 PSF)

CONFIGURATION 2 *Add Rotary Joint and Battery Modules*

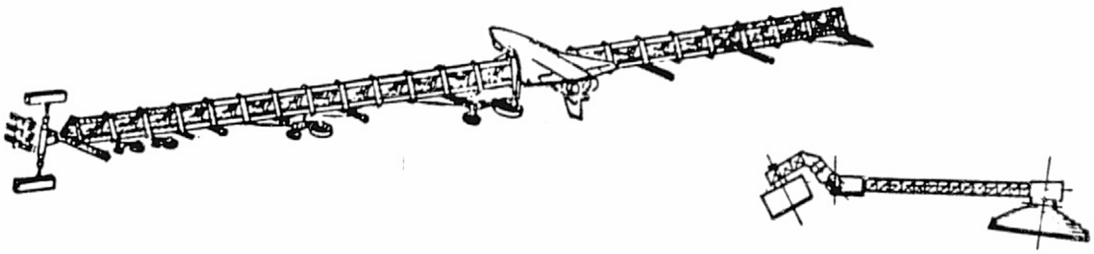
	WITH ORBITER	WITHOUT ORBITER
WEIGHT	103,682 KG (228,600 LB)	31,117 KG (68,600 LB)
$(W/C_DA)_{min}$	77.6 KG/M² (15.9 PSF)	46.8 KG/M² (9.6 PSF)
$(W/C_DA)_{max}$	204 KG/M² (41.8 PSF)	71.2 KG/M² (14.6 PSF)
$(W/C_DA)_{avg}$	100 KG/M² (20.5 PSF)	53.7 KG/M² (11.0 PSF)

Figure 5.1-6. Platform Construction Configurations 1 and 2

CONFIGURATION 3 *Add Solar Array Canisters and 2 RCS Modules*

	WITH ORBITER	WITHOUT ORBITER
WEIGHT	115,475 KG (254,600 LB)	122,097 KG (269,200 LB)
(W/C_DA)	81 KG/M^2 (16.6 PSF)	57.6 KG/M^2 (11.8 PSF)
(W/C_DA)	216.6 KG/M^2 (44.4 PSF)	93.2 KG/M^2 (19.1 PSF)
(W/C_DA)	1034 KG/M^2 (21.2 PSF)	65.4 KG/M^2 (13.4 PSF)

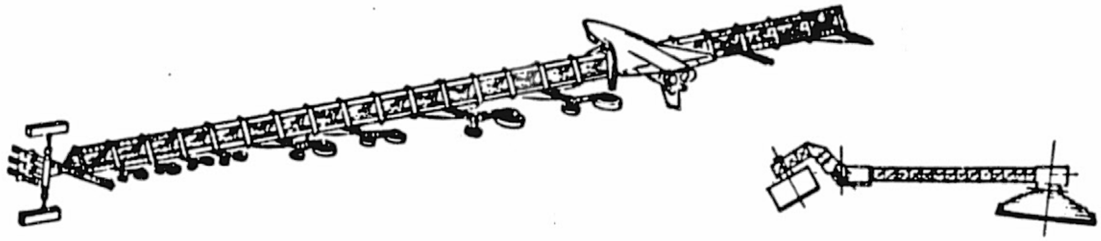
CONFIGURATION 4 *Add 8 Antenna Modules*



	WITH ORBITER	WITHOUT ORBITER
WEIGHT	122,097 KG (269,200 LB)	49,528 KG (109,200 LB)
(W/C_DA)	76.6 KG/M^2 (15.7 PSF)	53.7 KG/M^2 (11.0 PSF)
(W/C_DA)	178.1 KG/M^2 (36.5 PSF)	95.2 KG/M^2 (19.5 PSF)
(W/C_DA)	98.1 KG/M^2 (20.1 PSF)	62.5 KG/M^2 (12.8 PSF)

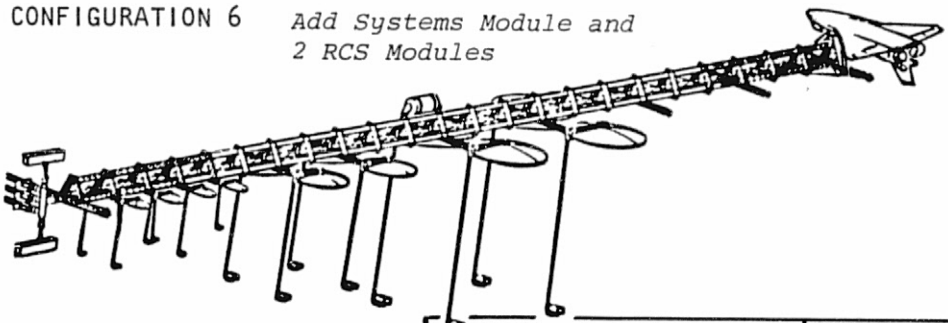
Figure 5.1-7. Platform Construction Configurations 3 and 4

CONFIGURATION 5 *Add 8 More Antenna Modules*



	WITH ORBITER	WITHOUT ORBITER
WEIGHT	128,719 KG (283,800 LB)	56,150 KG (123,800 LB)
$(W/C_DA)_{max}$	72.7 KG/M² (14.9 PSF)	96.5 KG/M² (19.8 PSF)
$(W/C_DA)_{min}$	172.2 KG/M² (35.3 PSF)	51.2 KG/M² (10.5 PSF)
$(W/C_DA)_{avg}$	93.2 KG/M² (19.1 PSF)	60.5 KG/M² (12.4 PSF)

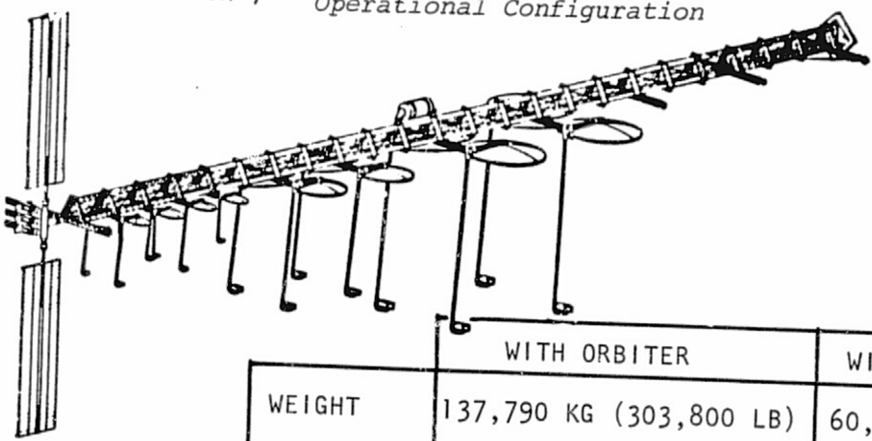
CONFIGURATION 6 *Add Systems Module and 2 RCS Modules*



	WITH ORBITER	WITHOUT ORBITER
WEIGHT	237,790 KG (303,800 LB)	65,221 KG (143,800 LB)
$(W/C_DA)_{max}$	75.1 KG/M² (15.4 PSF)	56 KG/M² (11.5 PSF)
$(W/C_DA)_{min}$	177.6 KG/M² (36.4 PSF)	106.9 KG/M² (21.9 PSF)
$(W/C_DA)_{avg}$	96.1 KG/M² (19.7 PSF)	66.8 KG/M² (13.7 PSF)

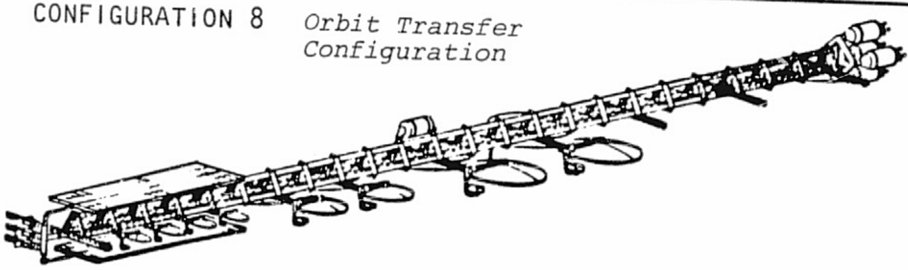
Figure 5.1-8. Platform Construction Configurations 5 and 6

CONFIGURATION 7 *Operational Configuration*



	WITH ORBITER	WITHOUT ORBITER
WEIGHT	137,790 KG (303,800 LB)	60,686 KG (133,800 LB)
$(W/C_DA)_{min}$		
$(W/C_DA)_{max}$		
$(W/C_DA)_{avg}$	35.1 KG/M^2 (7.2 PSF)	22.4 KG/M^2 (4.6 PSF)

CONFIGURATION 8 *Orbit Transfer Configuration*



	WITHOUT ORBITER
WEIGHT	60,685 KG (133,800 LB) 199,565 KG (440,000 LB)
$(W/C_DA)_{min}$	61.0 KG/M^2 (12.5 PSF)
$(W/C_DA)_{max}$	205.4 KG/M^2 (42.1 PSF)
$(W/C_DA)_{avg}$	133.2 KG/M^2 (27.3 PSF)

Figure 5.1-9. Platform Configurations for LEO Checkout and Orbit Transfer

The second post-construction phase was the orbit transfer propulsion build-up operation. During this build-up activity, five fully fueled propulsion modules would be delivered and installed on the completed communications platform. The five-module propulsion system is required for transporting the platform to its operational geosynchronous orbit position. The platform was assumed to be in a low drag configuration during this phase as shown at the bottom of Figure 5.1-9. The solar arrays would be folded back along the tri-beam structure in the position they would occupy for the orbit transfer maneuvers. The communications antenna system would be partially deployed, which is their orbit transfer configuration also. The reflector dishes would be open but the feed system masts would be retracted. The vehicle orientation would be with the long axis in the orbit plane and aligned with the velocity vector. As propulsion module weight is added to the configuration, the ballistic coefficient changes from W/C_DA = 61 kg/m^2 (12.5 psf) to 205.4 kg/m^2 (42.1 psf), with an average value of 133.2 kg/m^2 (27.3 psf). A propulsion delivery schedule of one module every 20 days was assumed.

The resulting overall decay profile is shown in Figure 5.1-10 for an initial altitude of 463 km (250 nmi). Both a nominal case and an extended case, allowing for a 60-day contingency between construction steps 5 and 6 are shown. During the assumed nominal construction period of 294 days, the orbit is shown to decay to 402 km (217 nmi). An additional 16 km decay increment to 386 km (209 nmi) is shown for the 60-day contingency extension case. Thus, even for a significantly longer contruction interval, 284 days including propulsion delivery—compared to 127 days for the SPS test vehicle—the decay increment for an initial construction orbit altitude of 463 km (250 nmi) is acceptable for the communications platform. The SPS test vehicle could not be constructed with confidence at this altitude. These data, then, illustrate the differences in orbit decay and construction orbit altitude that can occur due to variations in the configuration of the systems being constructed.

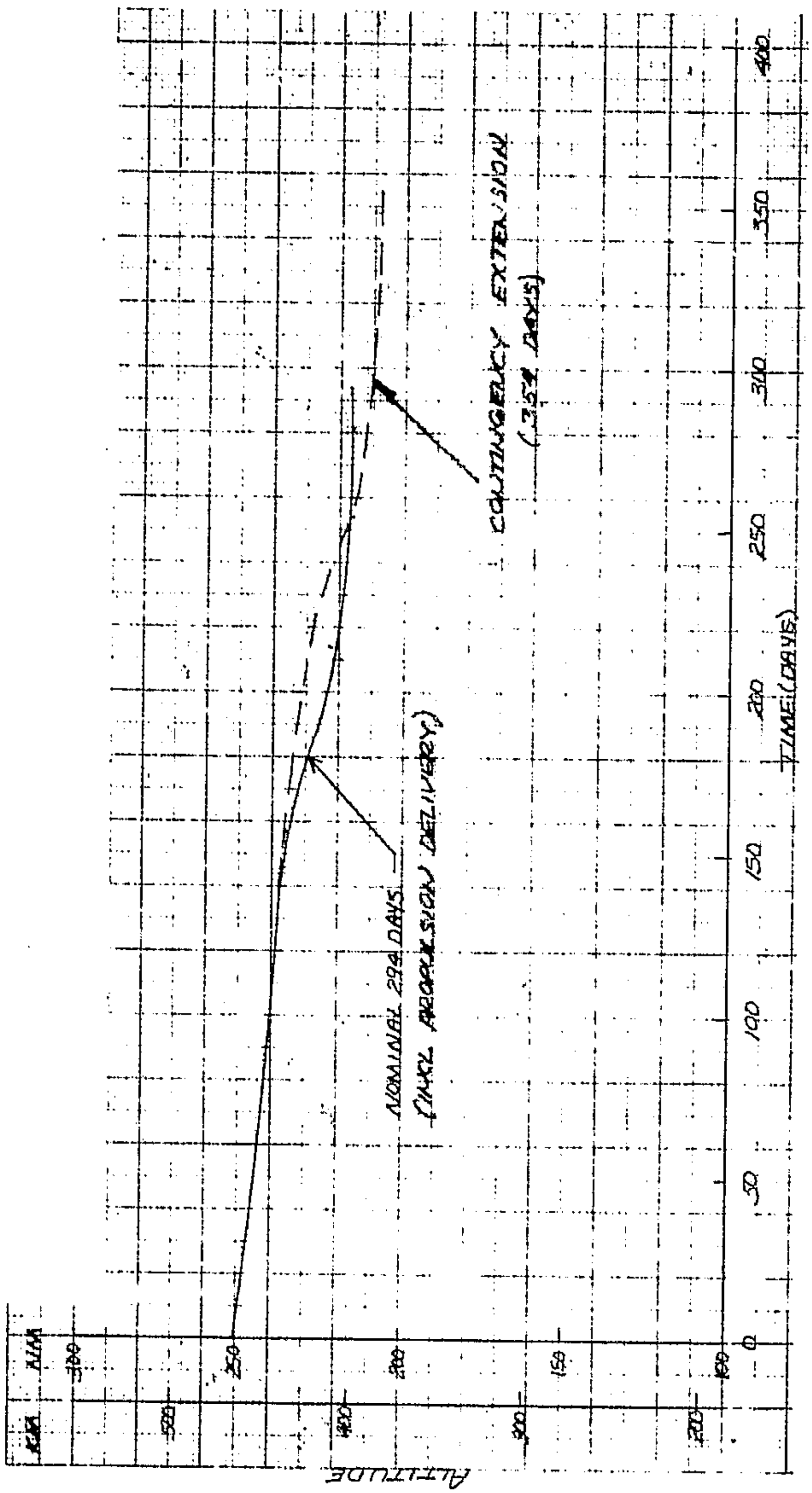


Figure 5.1-10. Orbit Decay During Construction of Advanced Communications Platform

5.1.2 Radiation Environment Impacts

The natural radiation environment is another factor which could potentially affect orbit altitudes for space construction. Of particular concern is the increased exposure during EVA activity and the possible need for designing automated construction techniques and processes which minimize the use of manned EVA participation. The nature of the radiation hazard and the preliminary orbit altitude limits for various EVA cases are briefly discussed below.

For 28° inclination circular orbits, only Van Allen belt electrons and protons are significant. Solar flare particles are excluded by the geomagnetic field (cutoff energies $\gtrsim 3$ GeV) and galactic (cosmic ray) particles contribute $\lesssim 10^{-2}$ rad/day independent of shielding (for $\lesssim 100$ gm/cm^2). In the absence of man-made nuclear radiation, therefore, only the Van Allen belts need be considered.

The Space Shuttle orbiter has an effective shield thickness for the crew of ~3 gm/cm^2. Thus, the cutoff energies are ~50 MeV for protons and ~5 MeV for electrons. For EVA operation the typical space suit provides ~0.2 gm/cm^2 effective shielding, which has cutoff energies of ~11 MeV (protons) and ~0.45 MeV (electrons). The particle fluxes of interest are those with energies above these cutoff energies.

Calculations have been carried out to obtain the Van Allen particle fluxes and tissue dose rates as a function of altitude for 28° inclination circular orbits. The SREP computer code was used to calculate the daily electron and proton fluxes, with the flux-to-tissue dose conversions being accomplished by hand calculations. The results obtained were tissue doses for 0.2 and 3 gm/cm^2 shielding as functions of altitude.

The Van Allen belt skin dose rates as a function of altitude are shown in Figure 5.1-11 for two shielding thicknesses—0.2 gm/cm^2 and 3 gm/cm^2. These are daily averages for circular orbits with an inclination of 28°. At the altitudes of interest ($\lesssim$1000 km) most of these doses will be received in the South Atlantic anomaly. Since the spacecraft passes through this anomaly only 3 to 7 orbits per day, depending upon altitude, it may be possible to schedule short-term ($\lesssim$6 hr) EVA during the orbits when the South Atlantic anomaly will not be encountered. No account of this effect (which is not important above ~1000 km altitude) was taken in this analysis.

While there are no "official" radiation dose limits for astronauts, the National Academy of Sciences recommendations are often used for mission analysis studies. These recommendations, listed in Table 5.1-2 were used on this study. For small shield thicknesses (e.g., an EVA suit) the skin dose limits are the overriding factor, but for large shield thicknesses usually the bone marrow dose limits determine the mission limit (duration or altitude). The tissue dose rates for the skin, eyes, and bone marrow are shown in Figure 5.1-12.

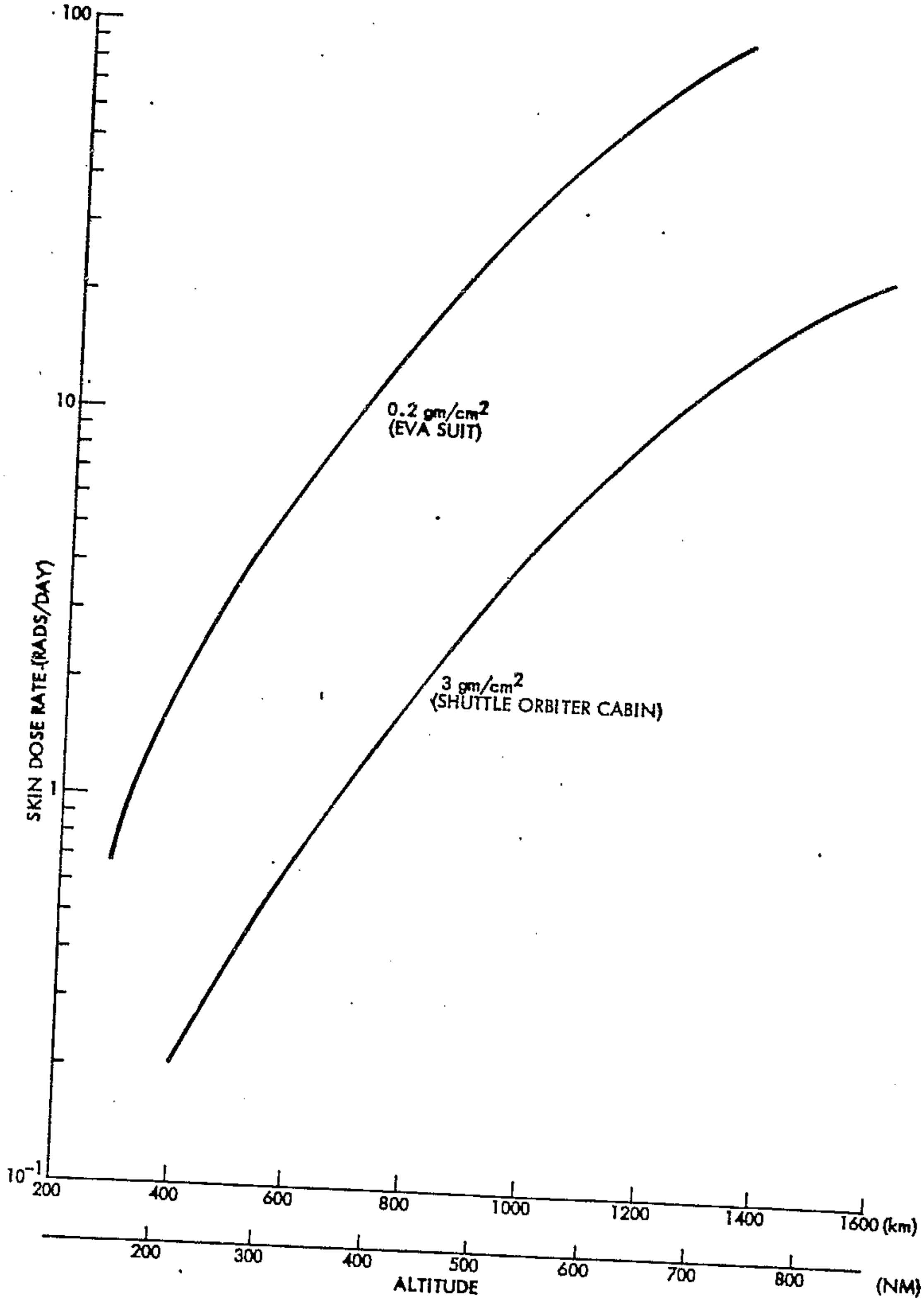


Figure 5.1-11. Skin Dose Rates in the Van Allen Belts

Table 5.1-2. Recommended Astronaut Dose Limits

Mission Duration	Dose Limit (rad) Bone Marrow (5 cm depth)	Skin (0.1 mm depth)	Eyes (3 mm depth)
30 days	25	75	37
90 days	35	105	52
180 days	70	210	104
1 year	75	225	112
Career limit	400	1200	600

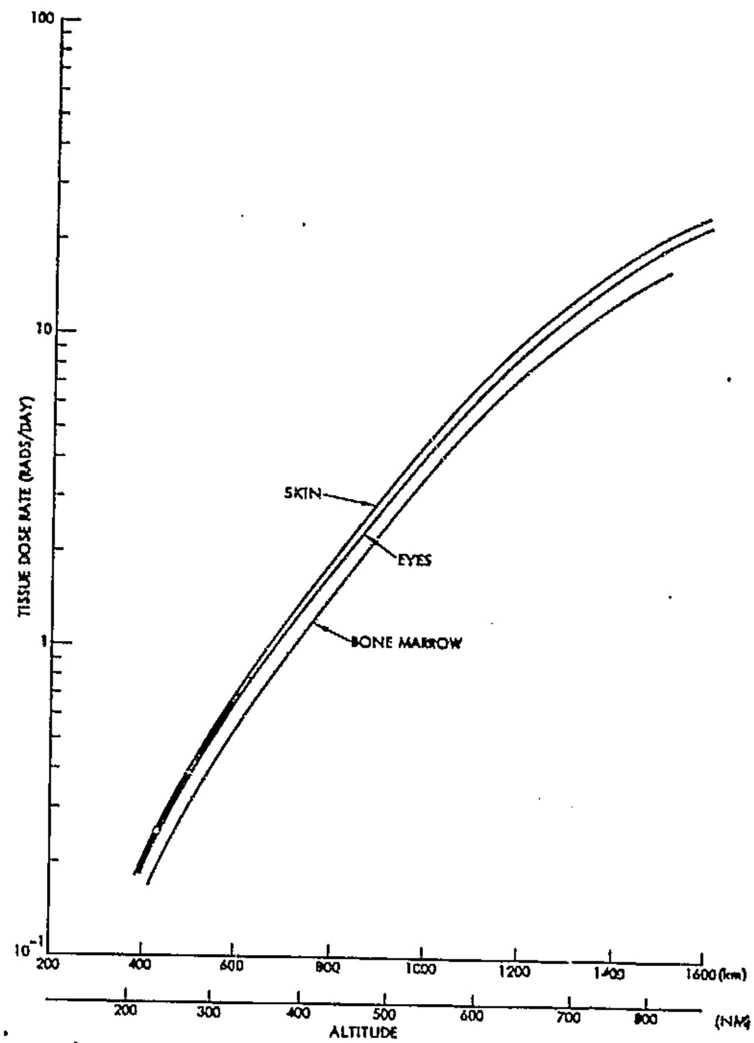


Figure 5.1-12. Tissue Dose Rates in the Van Allen Belts (Behind 3 gm/cm^2 Shielding)

In order to estimate the maximum altitude for the 7-, 10-, and 30-day missions, the 30-day dose limits of Table 5.1-2 were used. These are 75 rad (skin), 37 rad (eyes), and 25 rad (bone marrow). The numbers used to estimate maximum altitude for astronauts in the Shuttle orbiter cabin are listed in Table 5.1-3. The first part of the table lists the allowable dose rates (rad/day) permitted for the bone marrow, skin, and eyes. These numbers are merely the 25, 75, and 37 rad limits from Table 5.1-2 divided by the mission durations. The second part of the table lists the altitudes from Figure 5.1-12 for the dose rates in the top part of the table. It is seen that bone marrow is the limiting organ for all three mission durations.

Table 5.1-3. Maximum Altitude for Astronauts inside the Orbiter

Mission Duration (days)	Bone Marrow	Skin	Eyes
7	3.57	10.7	5.29
10	2.50	7.5	3.7
30	0.833	2.5	1.23

Mission Duration (days)	Maximum Altitude inside Orbiter (km)		
	Bone Marrow	Skin	Eyes
7	1000	1230	1060
10	920	1130	970
30	680	860	730

The corresponding numbers for EVA are listed in Table 5.1-4, except only the skin was used since it will be the limiting factor inside the 0.2 gm/cm^2 EVA suit. If continuous (24 hr/day) EVA were necessary, the maximum altitudes

Table 5.1-4. Maximum Altitudes for EVA Operation

Mission Duration (days)	Allowable Skin Dose Rate (rad/day)	Maximum Continuous EVA Altitude (km)
7	10.7	710
10	7.5	620
30	2.5	430

Mission Duration (days)	EVA Duration (days)	Allowable Skin Dose Rate (rad/day)	Maximum EVA Altitude (km)
7	1.75	43	940
10	2.5	30	860
30	7.5	10	620

(read from the top curve of Figure 5.1-11) vary from 800 km (7 days) to 430 km (30 days). However, by limiting EVA to 6 hr/day, the maximum altitudes can be increased to 1030 km (7-day missions) to 680 km (30-day missions). The numbers will be decreased somewhat to allow for the radiation doses received from the 18 hr/day when the astronauts are inside the cabin.

For example, on a 10-day mission, the astronaut will spend 2.5 days of EVA at a skin dose rate of ~10X and 7.5 days inside the cabin at a bone marrow dose rate of ~X. The total dose will be ~32.5X to the skin and ~10X to the bone marrow. (The bone marrow dose rate is approximtely the same for EVA and cabin occupancy.) Therefore the value of X for the skin is 75/32.5 = ~2.3, so that the EVA skin dose rate should be ~23 rad/day and the in-cabin dose rate to the bone marrow should be ~2.3 rad/day. These numbers lead to an EVA altitude of ~860 km (from Figure 5.1-11) and an in-cabin altitude of ~900 km (from Figure 5.1-2). To check, if the orbit altitude is the smaller of the two numbers (860 km), the skin dose rate will be 23 rad/day × 2.5 days = 57.5 rad during EVA and 2.5 rad/day × 7.5 days = 18.7 rad during cabin occupany (total 76.2 rad, slightly above the 75 rad allowed). The bone marrow dose will be ~2 rad/day × 10 days = 20 rad, less than the 25 rad allowed. This 860-km altitude is less than the 930 km allowed (on the basis of the EVA alone) or the 920 km allowed on the basis of cabin occupancy alone.

In this way (by iteration), the maximum altitudes for 7-, 10-, and 30-day missions with 25% EVA and 75% cabin (Shuttle orbiter) occupancy were calculated to be 940, 860, and 620 km, respectively. These are high enough that atmospheric drag will not unduly limit orbit lifetime. It is possible to increase the orbit altitudes somewhat by using a heavier space suit, but the difficulty of working in a heavier suit outweighs the slight orbit altitude increase. For example, increasing the EVA suit to 3 gm/cm^2 (the same shielding as the orbiter provides) would only increase the orbit altitude permitted by ~100 km.

If the orbit inclination were decreased to 0°, the environment decreases for orbit altitudes $\leq$ 900 km but increases for altitude $\geq$900 km (Figure 5.1-13). The effect of other orbit inclinations can also be seen. In the low altitude region of interest for space construction, the 30- to 60-degree orbit inclination band has the most severe radiation environment (due to the South Atlantic anomaly). Thus, the EVA altitude limits defined in the preceding tables are applicable to all orbit inclinations and, in fact, offer higher dose margins in the equatorial and solar inclination regions.

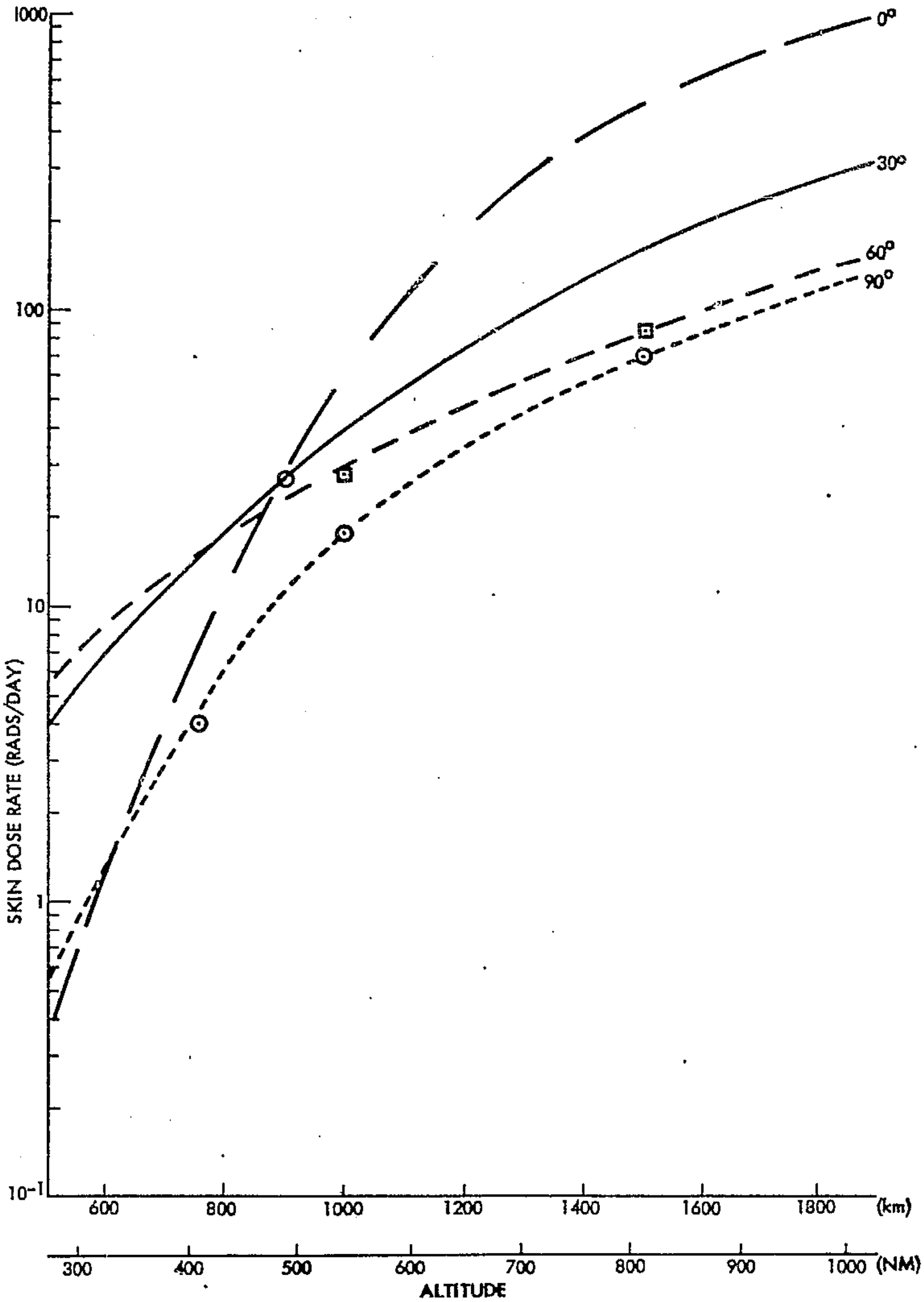


Figure 5.1-13. Effect of Orbit Inclination on EVA Skin Dose ($0.2\ gm/cm^2$ Shielding)

5.1.3 Orbiter Performance

In addition to orbit decay and radiation hazard, Shuttle delivery performance is the third major factor which must be considered in selecting suitable orbit altitudes for space construction. Shuttle payload capability is shown in Figure 5.1-14 as a function or orbit altitude. These data are for payload deliveries with rendezvous which would be typical for space construction of large area systems requiring more than one Shuttle flight. Superimposed on these performance curves are upper and lower altitude limits imposed by the radiation hazard on the high side and drag effects on orbit lifetime on the low side. Radiation hazard bounds are shown for both EVA and the orbiter crew cabin. Two limits for minimum altitude due to orbit decay are also shown. One corresponds to the higher decay rates associated with random unconstrained orientations and the other (lowest limit, ≈370 km) reflects low decay rates associated with orientation continuously constrained to maintain minimum drag values. The region between the "radiation" and "orbit decay" limits would be suitable for space construction.

This region tends to be centered on that portion of the Shuttle performance envelope which requires the inclusion of a single OMS kit. Thus, many construction payloads, depending upon packaging characteristics, would require the use of OMS. This would be particularly true for high drag configurations such as the SPS test vehicle. The intrusion of the OMS kit into the available cargo bay volume must therefore be considered in planning construction cargo manifests. Lower drag configurations might possibly be constructed at low enough altitudes that some of the construction flights, those with cargoes that are "volume limited", could be performed without the need for OMS. (Experience has indicated construction mission payloads tend to be volume-limited rather than weight-limited.)

Detailed analyses are required on the integrated construction process to adequately determine the actual drag history and orbiter bay packaging of the individual construction flights to more accurately determine the construction orbit altitude requirements for a given project system. These could be further refined by inclusion of solar cycle effects on atmospheric density for the projected project system schedules. However, the preliminary analysis reported here serves to identify the key factors affecting construction orbit altitude and highlights their significance to the specific project systems contained in the study.

5.1.4 Conclusions

The following principal observations/conclusions were derived from the preceding construction orbit analyses:

1. Shuttle payload performance and orbit decay due to lightweight/high-drag space construction configurations are the main drivers affecting construction orbit altitude.

2. Construction orbit altitude is project-dependent because orbit decay is affected by the configuration and construction time, both project-dependent.

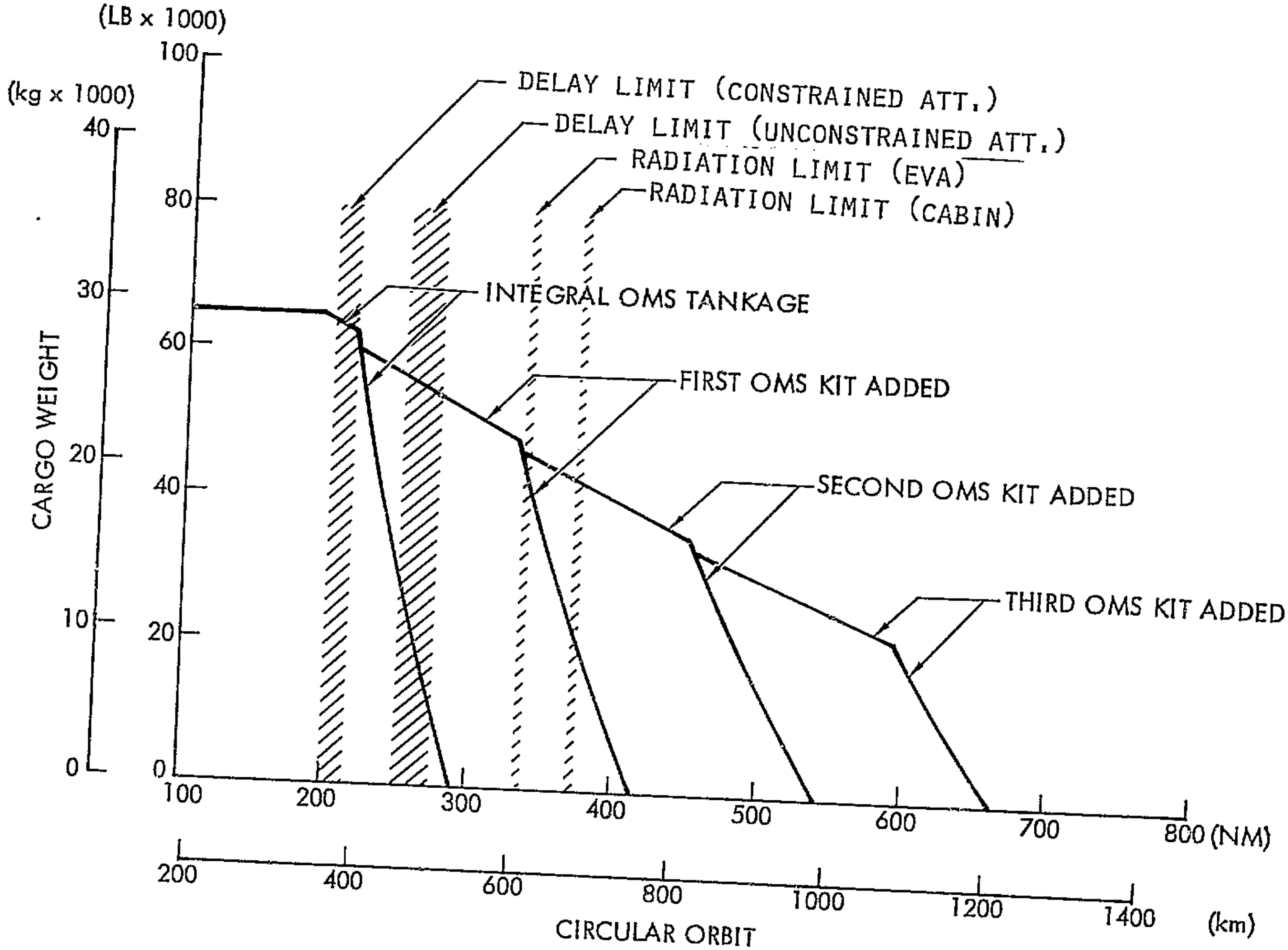


Figure 5.1-14. Orbit Altitude Limits for Space Construction

3. Large projects involving multiple Shuttle flights will likely require OMS kits to meet the minimum altitude limits for orbit decay.

4. Attitude control to minimize drag effects can significantly reduce minimum altitude limits for space construction, possibly to the point where OMS kits are not required.

5. The radiation hazard does not appear to be a driver on construction orbit altitude. Radiation altitude limits are significantly above the limits due to orbit decay. For construction orbit altitudes below 500 km (275 nmi), more than 100 EVA construction missions could be flown within career dosage limits.

6. The minimum construction orbit altitudes for the project systems considered in the study are:

   - SPS test vehicle, 510 km (275 nmi)
   - Advanced communications platform, 460 km (250 nmi)

   With drag control orientations (minimum drag), both projects could be constructed at altitudes as low as the 370-380 km range (200+ nmi).

7. Trades of orientation drag control vs. orbit makeup vs. high construction orbits are required to optimize the construction orbit altitude for each project system. An altitude range of 450-500 km (250-275 nmi) appears satisfactory for use in initial analyses of most projects.

5.2 ORBIT TRANSFER

Sizing analyses presented later in this section have shown propulsion system weights to greatly exeed the weight of the systems to be transported. For LEO-to-GEO orbit transfers, propulsion weights can range from 2-1/2 to 6 or 7 times the transported weight for chemical propulsion systems. For solar electric propulsion concepts, very large solar arrays will be required. Thus, orbit transfer propulsion can represent major fractions of the logistics requirements for large area space systems and will pose significant challenges to the build-up, assembly, and integration of propulsion systems into space constructed systems.

In addition to size there are several other major factors of concern in the integration of propulsion systems with the new types and scope of space systems which are designed to be constructed in space. These include concentrated thrust loads and T/W effects, structural stiffness interactions with thrust vector control, techniques for the delivery of propulsion modules/sytems to the space construction site, and the special implications associated with solar array size and LEO-to-GEO trip times attendant with solar electric propulsion concepts.

Thus, the orbit transfer analysis presented here includes technology/sizing considerations for various types of propulsion systems along with preliminary assessments of the other main integration issues. Emphasis is on the identification of drivers which can have significant impacts on the construction and design of large area space systems rather than on propulsion optimizations.

The space-fabricated tri-beam configuration for an advanced communications platform (Figure 5.1-9) was used as the reference configuration for comparing various advanced propulsion concepts. This project system is 230 m long and weighs approximately 61,000 kg (134,000 lb).

5.2.1 Advanced Cryogenic Propulsion Concept

The advanced cryogenic propulsion concept used in the study utilizes liquid oxygen (LO_2) and liquid hydrogen (LH_2) propellants at a mixture ratio of 6 in a cluster of high-pressure, staged-combustion engines (Figure 5.1-1).

5.2.1.1 Technology Considerations

Types of cryogenic propellant engines may be categorized to include the following engine cycle descriptors: expander, gas generator, staged combustion, and hybrid (plug cluster, etc.). The performance variations in terms of specific impulse with LO_2/LH_2 varies from 444 sec to 473 sec, depending upon the particular engine cycle, chamber pressure, and nozzle expansion ratio. For this study, a scaled-down staged-combustion cycle engine was selected based on the Rocketdyne 20 Klb thrust (90K newton) Advanced Space Engine (ASE) in technology development for NASA-LeRC. The characteristics of the scaled version used herein are summarized in Table 5.2-1.

Table 5.2-1. Engine Performance Summary

Thrust	22,240 N (5000 lb)
Chamber pressure	10,342 kPa (1500 psia)
Nozzle expansion area	400:1
Propellants	LO_2/LH_2
Mixture ratio, O/F	6:1
Specific impulse	4580 N-sec/kg (467 sec)
Overall length	1.32 m (52 in.)
Nozzle exit diameter	0.76 m (30 in.)
Weight	49.9 kg (110 lb)

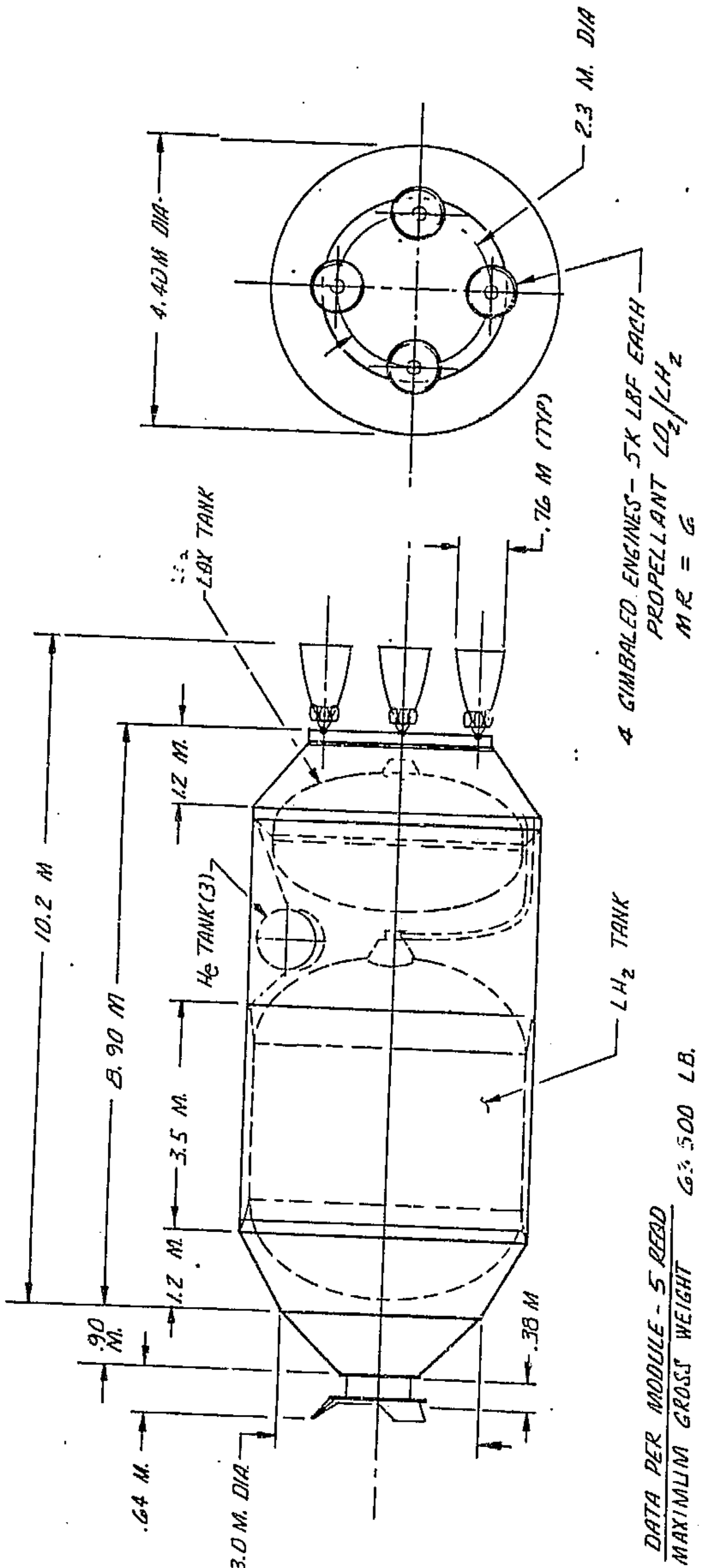
The 22,240-N (5000-lb) engine thrust level allowed the use of a multiple engine installation which provided (1) flexibility in controlling thrust-to-weight (T/W) by sequential engine shutdown (step-throttling), (2) thrust vector control (TVC) by two-axis gimbaling of multiple engines installed on multiple propulsion modules, and (3) shorter engine lengths without the need for nozzle retraction provisions. This flexibility was obtained at the expense of some performance, since this scaled-down specific impulse was reduced from the full-scale ASE I_{sp} of 473 sec.

Another technology consideration affecting design and performance was the stage mass fraction of the low-thrust propulsion module. A stage mass factor of 0.879 was used, and it is defined as the propellant weight divided by the total stage weight. The stage weight includes the inert weight of sub-systems such as structure, thermal control, avionics, propulsion, residual fluids and contingencies. The mass fraction value used stemmed from current OTV design studies which are based on many previous design studies including the NASA Tug and USAF Orbit-to-Orbit Shuttle (OOS).

The current propulsion module is shown in Figure 5.2-1 with some details and overall dimensions. A single oxidizer tank, fuel tank, and helium pressurization gas tanks are located within a structural shell that acts as a micrometeoroid shield. The design features the use of non-integral propellant tanks with multi-layer insulation and fiberglass tank supports for control of boiloff.

5.2.1.2 Overall Sizing

The propulsion module was sized to take full advantage of the current Space Shuttle orbiter payload capability and to minimize the number of orbiter flights and operational costs, since multiple modules are indicated from the magnitude of the platform weight involved. A maximum gross weight for the orbit transfer propulsion module of 28,800 kg (63,500 lb) was established. The required number of modules is determined by the platform weight requirements, the velocity increments for orbit transfer, and propulsion specific impulse values. Propellant off-loading can be used in matching the platform weight requirements with the basic module. A platform weight of 60,500 kg (133,400 lb) is used as an example for the following discussions.



DATA PER MODULE - 5 REQD	
MAXIMUM GROSS WEIGHT	6[illegible] 500 LB.
MAXIMUM PROPELLANT LOAD	54 815 LB.
EMPTY WEIGHT	7685 LB.
STAGE MASS FRACTION	0.879

Figure 5.2-1. Low Thrust Propulsion Module

The propulsion module design weight summary with maximum propellant loading is shown in Table 5.2-2.

Table 5.2-2. LTP Maximum Propellant Load Conditions

	kg	(lb)
Maximum gross weight	28,803	(63,500)
Maximum propellant load	25,317	(55,815)
Inert weight	3,486	(7,685)
Stage mass fraction	0.879	
6% propellant boiloff	1,519	(3,349)
Usable propellant (after boiloff)	23,798	(52,466)

5.2.1.3 Boiloff Management

Propellant storability is a requirement for the entire elapsed time from propellant tanking to burnout. The use of cryogenic propellant requires adequate insulation for tanks to minimize boiloff propellant losses. Transit times to LEO and, subsequently, to GEO are relatively short (measured in hours), so that the elapsed time that impacts boiloff the greatest is the time required in LEO to accumulate the necessary number of propulsion modules. This elapsed time may be on the order of eight weeks, based on the following simplified scenario:

- Multiple flights are required to transport material and subsystem modules to LEO.
- The orbiter requires a two-week turnaround period between flights.
- A total of five propulsion modules are required, thus requiring four two-week periods between the delivery of the first and fifth modules.

From this example, it can be seen that the fifth module arrives in LEO eight weeks after the first module. Contingencies and margin allowances may be accounted for by assuming all modules have eight weeks of propellant boiloff, although only the first module has experienced the entire eight-week holding period.

Insulation concepts from prior studies (Ref. 2, 3, and 4) of LO_2/LH_2 propulsion modules include the use of multi-layer insulation externally applied to non-integral propellant tanks that are supported within an outer shell by fiberglass struts that act as heat blocks. Insulation materials such as layers (3/4 to 1 inch total thickness) of double aluminized Mylar and use of fiberglass tank supports will limit boiloff rates to 1.04 kg/hr (2.3 lb/hr) for tanks designed to contain 25,400 kg (56,000 lb) of LO_2/LH_2. This results in boiloff rates of 0.7% per week with 120 kg (265 lb) of insulation. For the eight-week holding period, a boiloff allowance of 6% was used with the above insulation concept. Alternative insulation materials, such as layers of double goldized Kapton, would provide lower boiloff rates and less insulation weight for an increase in material costs. This latter concept may prove more desirable for single module applications with reusable requirements.

5.2.1.4 T/W Trade

The T/W impact on vehicle weight and the effects of T/W on delta-V requirements enter a tradeoff relationship that was investigated during the study. In addition, the T/W effect on engine burn time requirements is also shown.

Delta-V Vs. T/W

The effect of T/W on delta-V is shown over a wide range of values in Figure 5.2-2. For application to the cryogenic propulsion module, a narrower range of T/W is of more interest, such as that shown plotted to an expanded scale in Figure 5.2-3. It is recognized that the single perigee burn data accentuates the effect of T/W values less than 0.1, compared to multiple burn effects; nevertheless, the trend would remain the same—that is, a marked increase in velocity increment is required at values less than 0.1 g.

In addition, it should be noted that for determining the propellant weight requirements, the delta-V requirements used here are expressed in terms of an average T/W value. "Average" refers to an average between the initial and burnout conditions, whereas, the maximum T/W value (usually at burnout conditions) is of interest to the structural designer. A correlation between average and maximum T/W values for a typical platform system is illustrated in Figure 5.2-4, which also relates this correlation to generic types of engine thrust control. Variable throttling to low levels would be required if a constant T/W is needed. The maximum T/W value will determine the thrust requirements at burnout, and impact engine requirements for either the use of multiple engines with sequential shutdown to control T/W as propellants are consumed, or the use of fewer engines but with throttling requirements for T/W control.

Weight Vs. T/W

In addition to the aforementioned delta-V impact of T/W, weight penalties will also grow for large space systems if the T/W is too great. A small regime of appropriate T/W is indicated, to avoid both the propellant and system weight penalties of high delta-V (T/W too low) and the structural weight penalty of too large a T/W value. This is illustrated for a large advanced communications satellite (fabricated of graphite composite material) in Figure 5.2-5, which shows the satellite weight as a function of the maximum design T/W for two types of structure. When this T/W effect is combined with the delta-V effect, an optimum T/W is indicated at 0.28, as shown in Figure 5.2-6, for the erectable pentahedral truss-type structure. A maximum design T/W of 0.2 was selected for this particular spacecraft. The curve is relatively flat between T/W values of 0.2 and 0.4 g. A total of five propulsion modules is also indicated from the figure. Six would be required with T/W's below approximately 0.12.

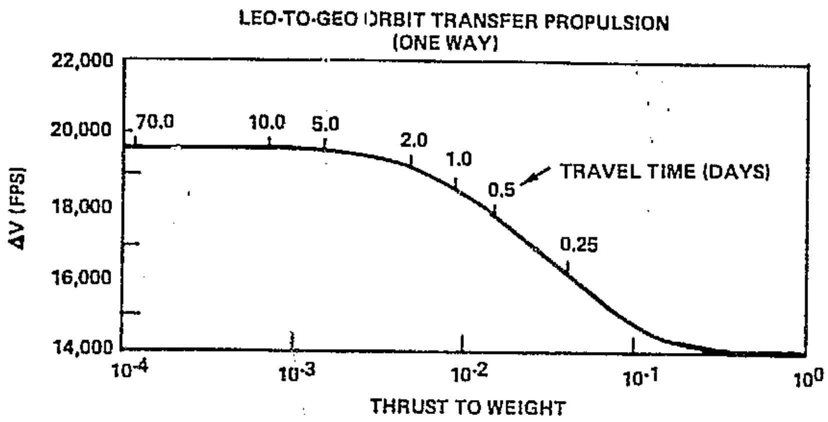


Figure 5.2-2. Delta-V Requirements Vs. T/W Ratio

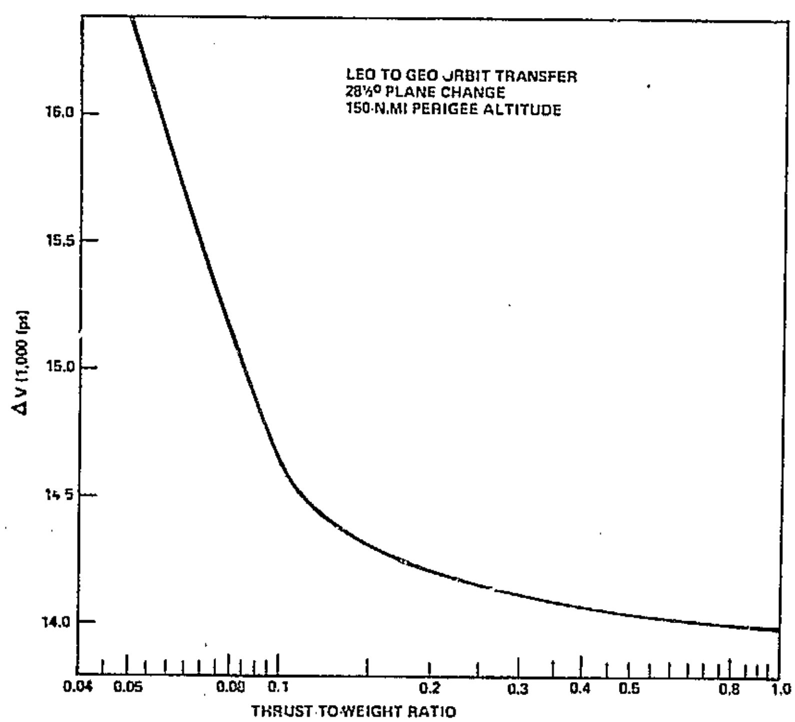


Figure 5.2-3. Expanded Scale of ΔV Requirements Vs. T/W Ratio

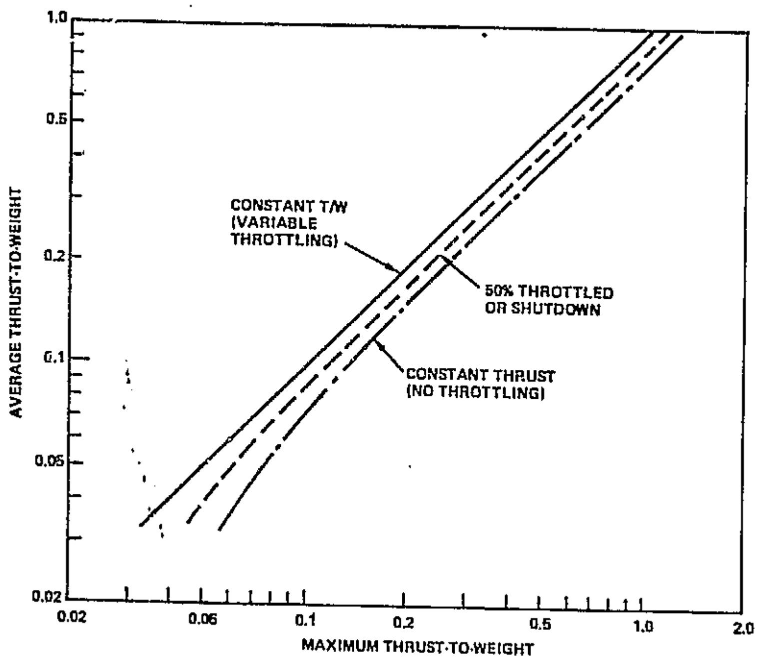


Figure 5.2-4. Maximum Vs. Average T/W Ratio Throttling Effects

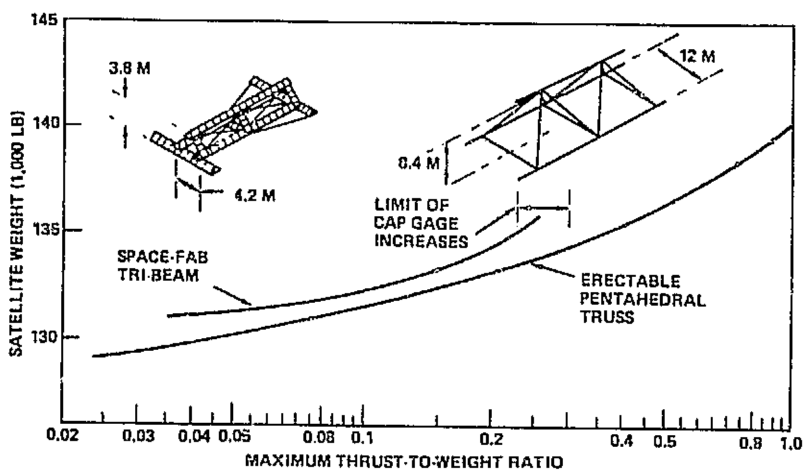


Figure 5.2-5. Large Space System Weight Vs. Maximum T/W Ratio

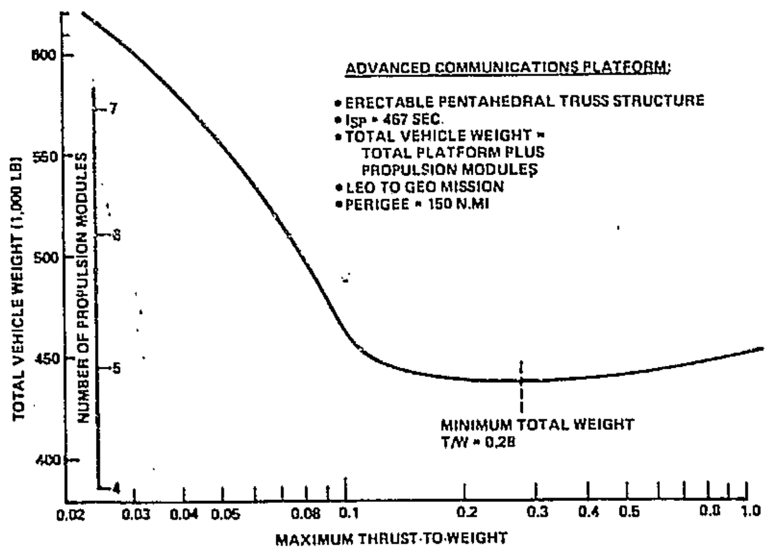


Figure 5.2-6. T/W Ratio Impact on Total Vehicle Weight

Burn Time Vs. T/W

The burn time of low-thrust engines operating on long missions at low T/W ratios will impose a new technology requirement for future space engine developments. For example, the effect of T/W on engine burn time for the example LEO-to-GEO one-way mission is illustrated in Figure 5.2-7. At T/W values of 0.1 to 0.2, burn times of 1200 to 2200 seconds are required. Thus far, burn durations per mission of this magnitude have not been required for bi-propellant thrusters, although lower performance, low-thrust, mono-propellant thrusters have fired for hours, such as for the ATS-6 satellite. The actual burn times, including effects of throttling or engine shutdown, would be increased somewhat over those shown on the figure, which was based on a simplified approach. This example, however, does illustrate how burn time is affected by mission-related parameters of T/W and delta-V.

From a feasibility viewpoint, the actively cooled engine concepts being considered have essentially an unlimited burn time capability. Active cooling concepts for staged combustion engines include combinations of dump cooling, regenerative cooling, and transpiration cooling with the hydrogen fuel.

The primary impact of long burn time design requirements would be on development costs, and possibly weight. Weight growth during developmental phases could result from the accumulation of tolerances and margins allowed in design in those areas where long-duration experimental data are unavailable. Also, the developmental costs could be impacted by the developmental and qualification testing required to demonstrate the long burn time engine design.

5.2.1.5 Throttling Considerations

A variable thrust level may be desirable for limiting the burnout acceleration, (T/W), g-level for lightweight, flexible large space structures. Throttling may be accomplished by variable throttling a single engine by limiting the propellent flow rate and reducing the chamber pressure to operate at a reduced thrust level, such as 50 percent of full thrust. Throttling may also be accomplished by step-throttling or the sequential shutdown of a number of fixed thrust engines. These considerations were mentioned earlier in discussions of technology considerations and delta-V versus T/W.

In addition, throttling techniques have potential use in minimizing structural load amplification in the engine starting transient phase of thrust build-up. The rate of application of propulsive thrusting force to "soft" structural platform designs is of importance to avoid excess weight penalties. The thrust rise time during the engine starting transient must be long enough to keep the structural load amplification factor as close to a value of one as possible, since a value of 2, for example, could result in a 25% to 40% structural weight increase of the vehicle. The interaction between the engine thrust rise time and the amplification of structural loading is shown in Figure 5.2-8. Amplification factors varying from 1 to 2 can occur, depending on the shape of the thrust/time relationship (see inset), and the ratio of the period of thrust buildup, τ, to the period of the structure, T. The structural period, T, is associated with the lowest modal frequency. The lowest amplification factor results from a linear thrust rise shape, a = 0.

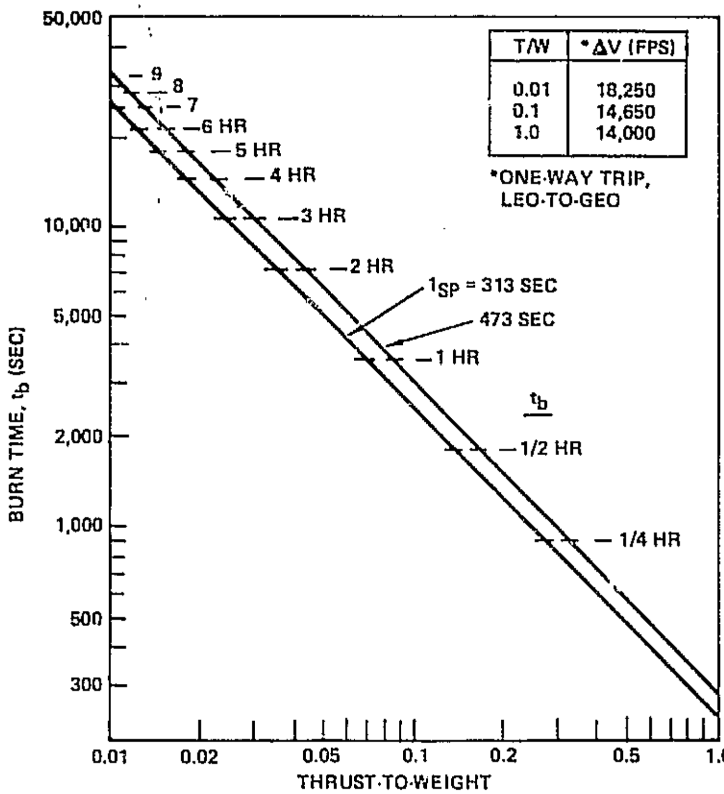


Figure 5.2-7. T/W Ratio Effect on Engine Burn Time

For a linear buildup of thrust in time, τ, and with the function τ/T greater than 2, the amplification factor will not exceed a value of 1.2

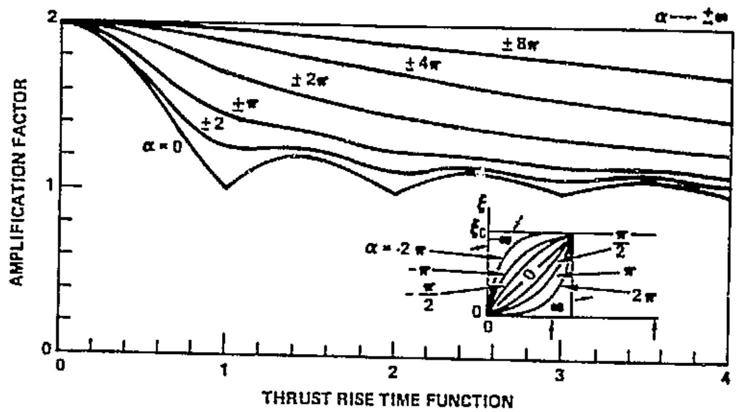


Figure 5.2-8. Thrust Rise Time Effects on Amplification Factor

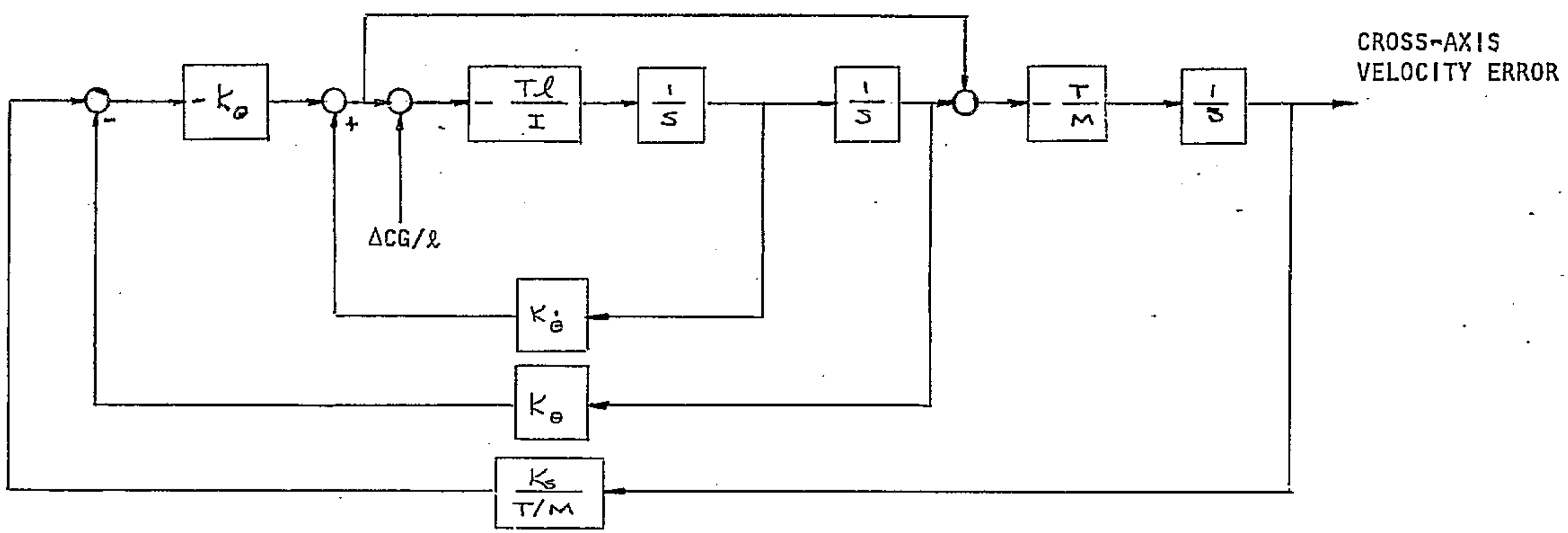
The implications of these relations for the propulsion system are that deep throttling and/or multiple low-thrust engines are required. Additional study effort is required in this technical area to define the engine starting transients required to avoid excessive load amplification, and determine starting sequences for multiple fixed thrust engines, and define the variable throttling capability of pump-fed and pressure-fed engines.

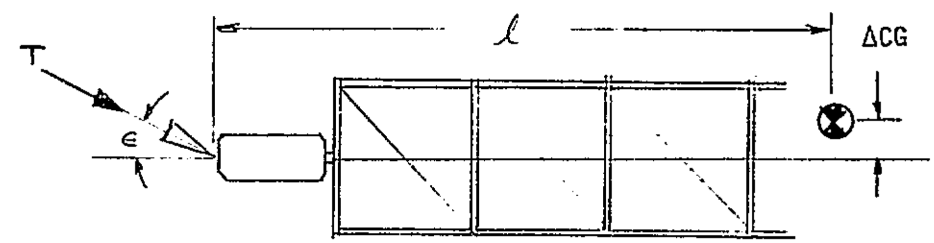
5.2.1.6 Thrust Vector Control/Structural Stiffness

A linear analysis of a conceptual, single-axis, thrust vector control system was made to determine potential velocity penalties at the end of orbit transfer as a function of control system frequency and center-of-gravity offset. Two control systems were considered. The first contains rate plus position feedback and the second adds to this a steering loop. The block diagram is shown in Figure 5.2-9. The gains were selected for a critically damped system and the T/W used was assumed to be a constant value of 0.17.

The total velocity penalty, or root sum square of the in-line and cross-axis components, as a function of control frequency is shown on Figure 5.2-10. The addition of the steering loop reduces the control frequency for a given velocity penalty or dramatically reduces the velocity penalties for a given control frequency.

It is desirable to separate the control frequency from the lowest structural frequency, which is also shown (Figure 5.2-10). For a control frequency 1/5 or 1/10 the structural frequency, which is sufficient separation to assure stable control, the system with steering provides a much lower velocity penalty or larger center-of-gravity offset. As shown in the figure, the total velocity penalty for the example communications platform configuration is of the order of 1 to 3 meters per second for a total LEO-to-GEO transfer delta-V of 4300 meters per second. Thus, control frequencies can be safely reduced to uncouple the control dynamics from the structural bending modes and adequate stiffness can be designed into large area space systems without significant affects on





$K_{\dot\theta}$ = RATE GAIN
K_θ = ATTITUDE GAIN
K_S = STEERING LOOP GAIN

Figure 5.2-9. TVC Systems Block Diagram

either delta-V propellants or structural weight. Detailed structural analyses of loads, stiffness, and bending modes for the reference configuration used here are presented in Section 3.3.2 of the Task 1 final report (SSD 79-0077).

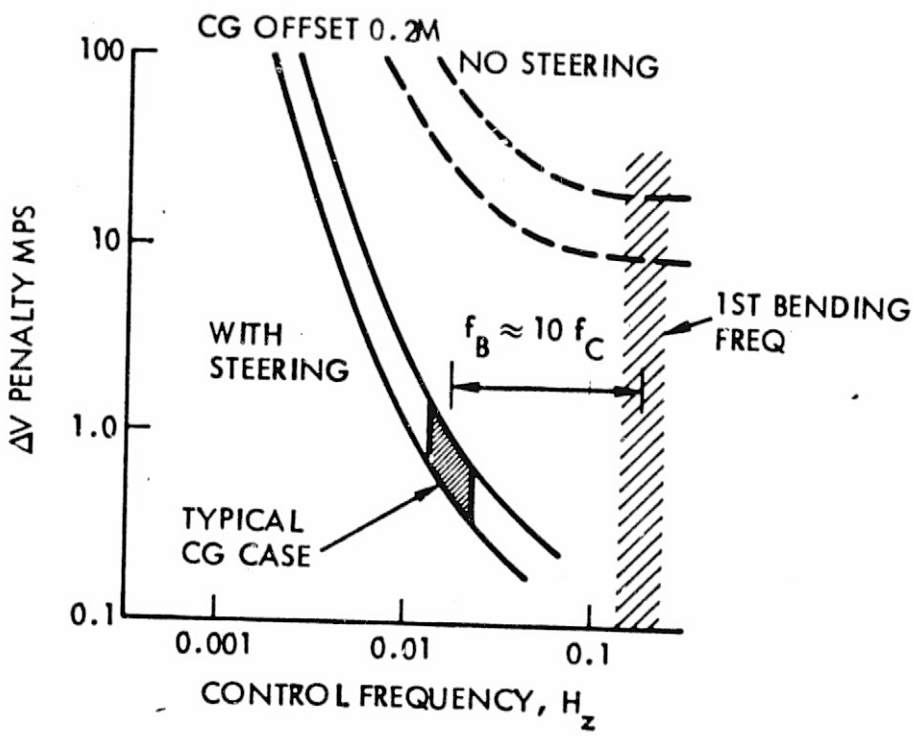


Figure 5.2-10. Velocity Penalty for
Low Control Frequencies

5.2.1.7 Alternate Propulsion Delivery Concepts

Once the large structure platform is completed in the Low Earth Orbit (LEO) the propulsion modules that will eventually boost the platform to an equatorial geosynchronous orbit (GEO) must be delivered in a safe and efficient manner. Atmospheric drag considerations can limit the platform construction altitude to altitudes above 300 nmi. From this altitude approximately 144,000 kg (317,500 pounds) of cryogenic propulsion modules are required to boost a 60,500 kg (133,000 lb) platform to geosynchronous orbit.

There are three fundamentally different techniques for delivering the propulsion modules to the user system (Figure 5.2-11). The first and most straight forward method is to deliver the propulsion modules by direct Shuttle Orbiter flights to the construction altitude. The second technique is one that uses the propulsion modules themselves to propel and dock the module to the constructed payload platform. This technique is called the "self" delivery technique. And finally, the third concept involves the use of a "teleoperator" concept to either deliver the propulsion modules to the construction site or bring the fully constructed payload platform down to an altitude that maximizes Shuttle payload delivery.

Direct Shuttle Delivery Concept

The direct Shuttle delivery of the propulsion modules to the construction orbit represents what might appear to be the easiest and simplest technique to implement. The Orbiter already incorporates all the necessary capability that is required for orbital transfer, rendezvous, and docking. There would be no need to develop a "teleoperator" vehicle nor an additional "intelligent" module to perform these tasks. All orbit transfer, rendezvous, and docking maneuvers would be performed by the manned Orbiter.

The Space Shuttle capability for payload delivery with and without rendezvous for orbits up to 650 km (350 nmi) altitude is shown in Figure 5.2-12. For the direct Shuttle delivery concept the lower values, i.e., delivery with rendezvous, must be considered. In this mode, with only the integral OMS tankage the maximum payload of 29,500 kg (65,000 lb) can be delivered only up to approximately 350 km (190 nmi). Drastic delivered payload reduction as a function of altitude is shown for altitudes above 403 km (218 nmi). The addition of one OMS kit reduces the usable payload bay length from 18.3 meters to approximately 15 meters. However, 21,800 kg (48,000 lb) of payload (including necessary cradles) can still be delivered to an altitude of 610 km (330 nmi). To facilitate rapid comparison of the various propulsion module delivery techniques this altitude (610 km) will be assumed to represent the construction orbit throughout the remainder of the propulsion module delivery analysis.

For the above conditions and assuming a reasonable cradle weight, nearly seven Shuttle flights would be required to deliver the 144,000 kg (317,500 lb) of propulsion modules. Each of the seven propulsion modules would then weigh only 20,600 kg (45,400 lb) allowing up to 1,180 kg (2,600 lb) for the cradle.

Orbiter payload center-of-gravity envelope requirements that also must be met are shown in Figure 5.2-13.

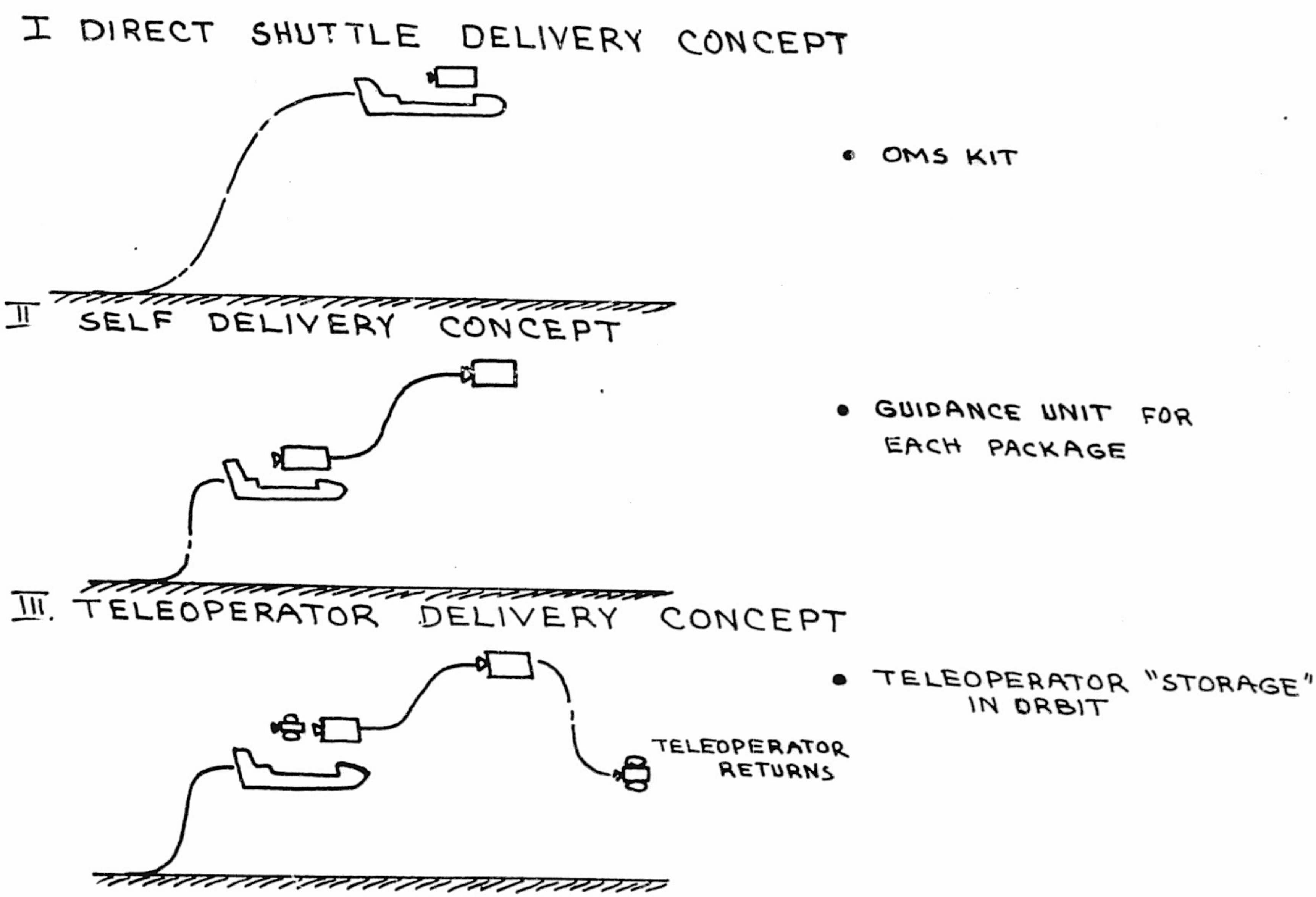


Figure 5.2-11. Delivery Concepts

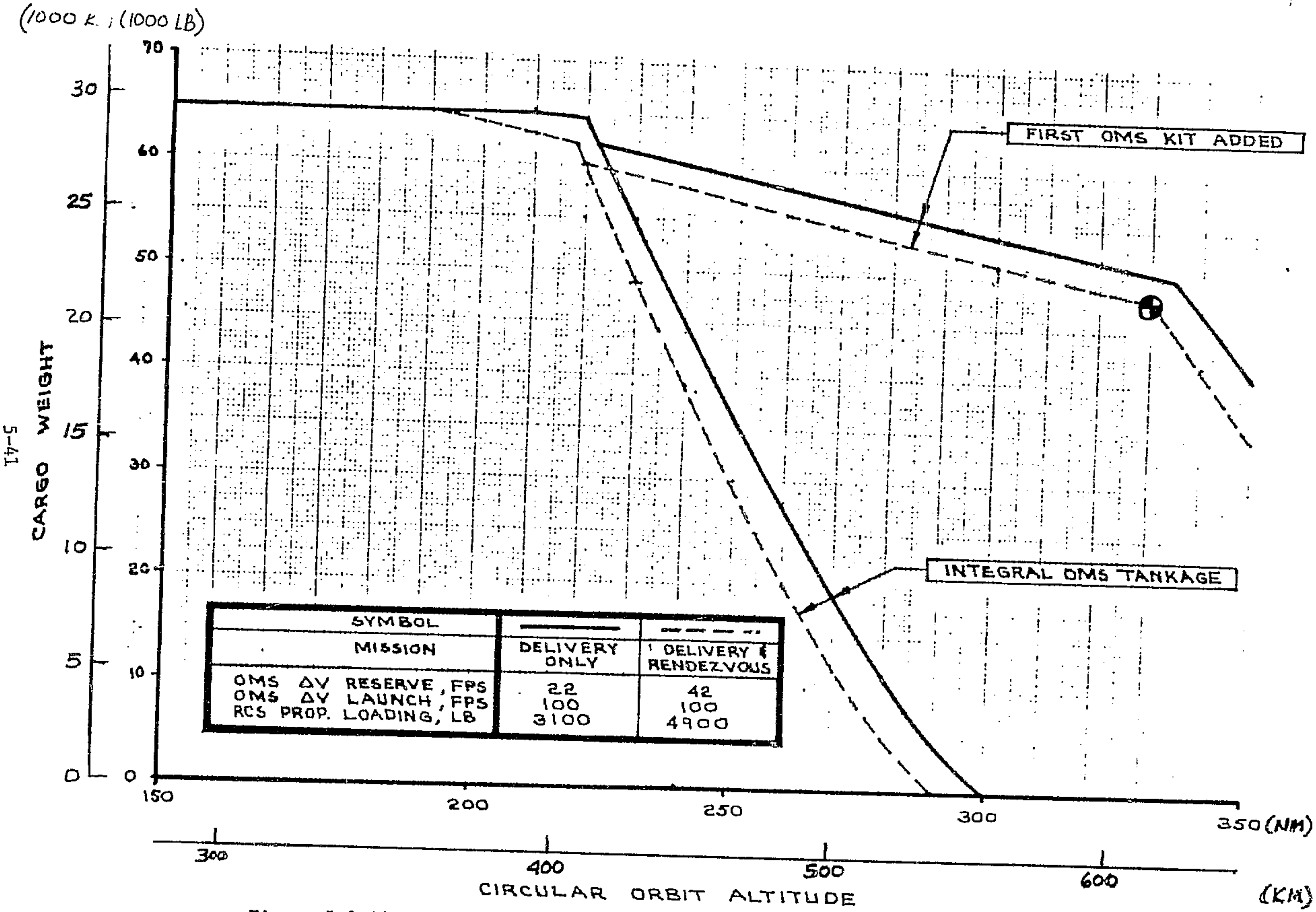


Figure 5.2-12. Space Shuttle Cargo Weight KSC Launch (Inclination = 28.5 Deg.)

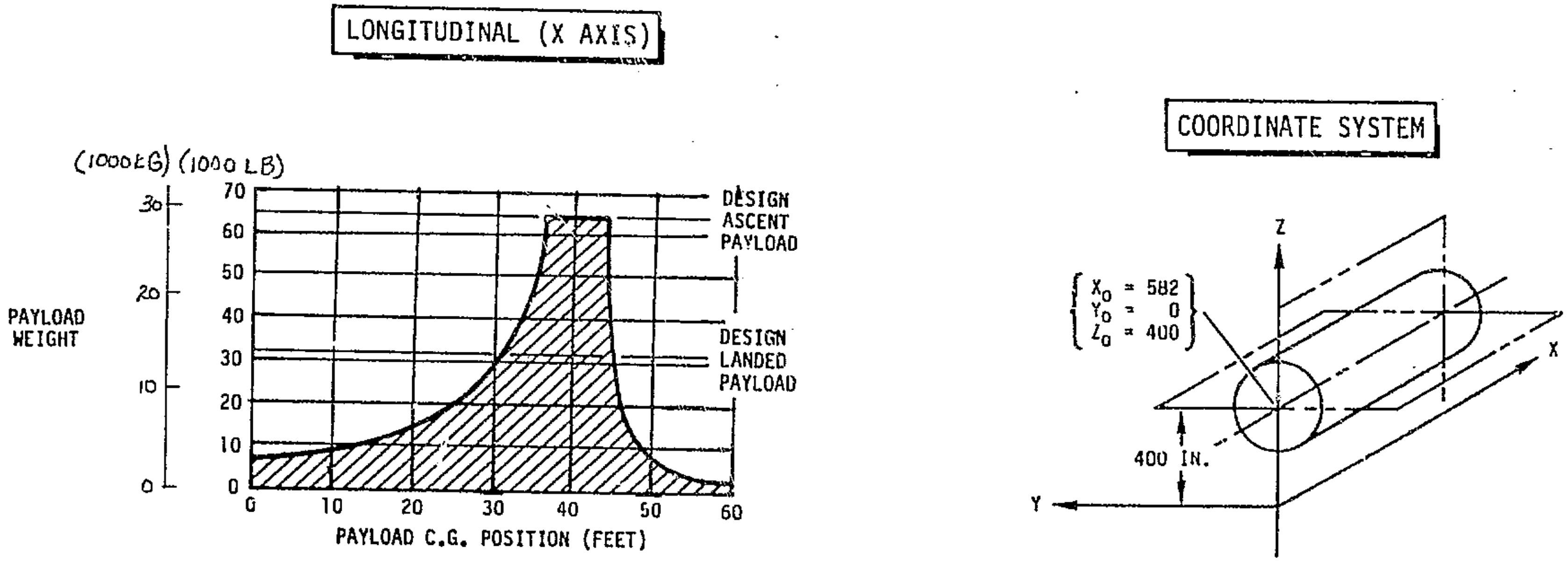


Figure 5.2-13. Payload Center-of-Gravity Envelope Requirements

C-3

The Self Delivery Concept

For the self delivery concept, as illustrated in Figure 5.2-11, the Shuttle delivers the maximum payload (delivery only - no rendezvous) to an altitude of approximately 405 km (220 nmi). This is consistent with the Space Shuttle capability shown in Figure 5.2-12 or 5.2-14. At this altitude the propulsion module is deployed by the Orbiter for self-transfer to the construction altitude. The ΔV for this transfer maneuver to 610 km (330 nmi) is 125 mps (415 fps), which includes 15 mps (50 fps) for final rendezvous and docking. There are no return flight requirements for this delivery technique.

The self delivery concept takes advantage of the higher structural efficiency and much better specific impulse of the propulsion modules to gain an advantage over the direct Shuttle delivery concept. Also, the weight of the orbiter need not be carried to the construction orbit in this mode, thereby yielding additional delivery performance benefits. The weight that can be delivered to altitudes above 405 km by this technique is shown in Figures 5.2-14 and 5.2-15.

The effect of rendezvous and docking ΔVs up to 30 mps (100 fps) is minimal on the weight delivered throughout the altitude range considered (Figure 5.2-14). Even the effect of lower specific impulse is not serious. In Figure 5.2-15 it can be seen that for the self delivery concept, even using Isp's associated with gaseous H_2/O_2, approximately 390 sec, the weight delivered to the 610 km (330 nmi) construction altitude is still above 27,900 kg (61,600 lb). This weight, however must also include the additional weight of the guidance, docking and vernier ΔV unit that is necessary for each of the propulsion modules. This requirement, in addition to lowering the useful propellant weight of the propulsion module would impose additional development costs.

For the self delivery concept it appears that six Shuttle flights would be required to deliver the 144,000 kg (317,500 lb) of propulsion modules. Each of the six propulsion modules would then weigh only 24,000 kg (53,000 lb), allowing up to 3,900 kg (8,600 lb) for the necessary rendezvous capability.

The problems of platform structural integrity as a result of docking impact as well as plume impingement during the final rendezvous maneuvers must be considered with this delivery concept.

Teleoperator Delivery Concepts

The teleoperator delivery concept includes all techniques where an auxiliary independent propulsion stage (other than the payload propulsion modules themselves) is used for the orbit transfer, rendezvous, and docking maneuvers. Most often such a unit will be reusable and contain all the necessary "intelligence" that would be required to perform the orbit transfer, and more important, the delicate final rendezvous and docking maneuvers. The term "teleoperator" will be applied to all these types of units in general.

Four techniques were considered under the teleoperator delivery concept. The first of these mploys individual teleoperators (based on the Martin concept for Skylab reboost) to deliver each propulsion module separately. The second technique involves a teleoperator farm, where a single "intelligent" core unit may be

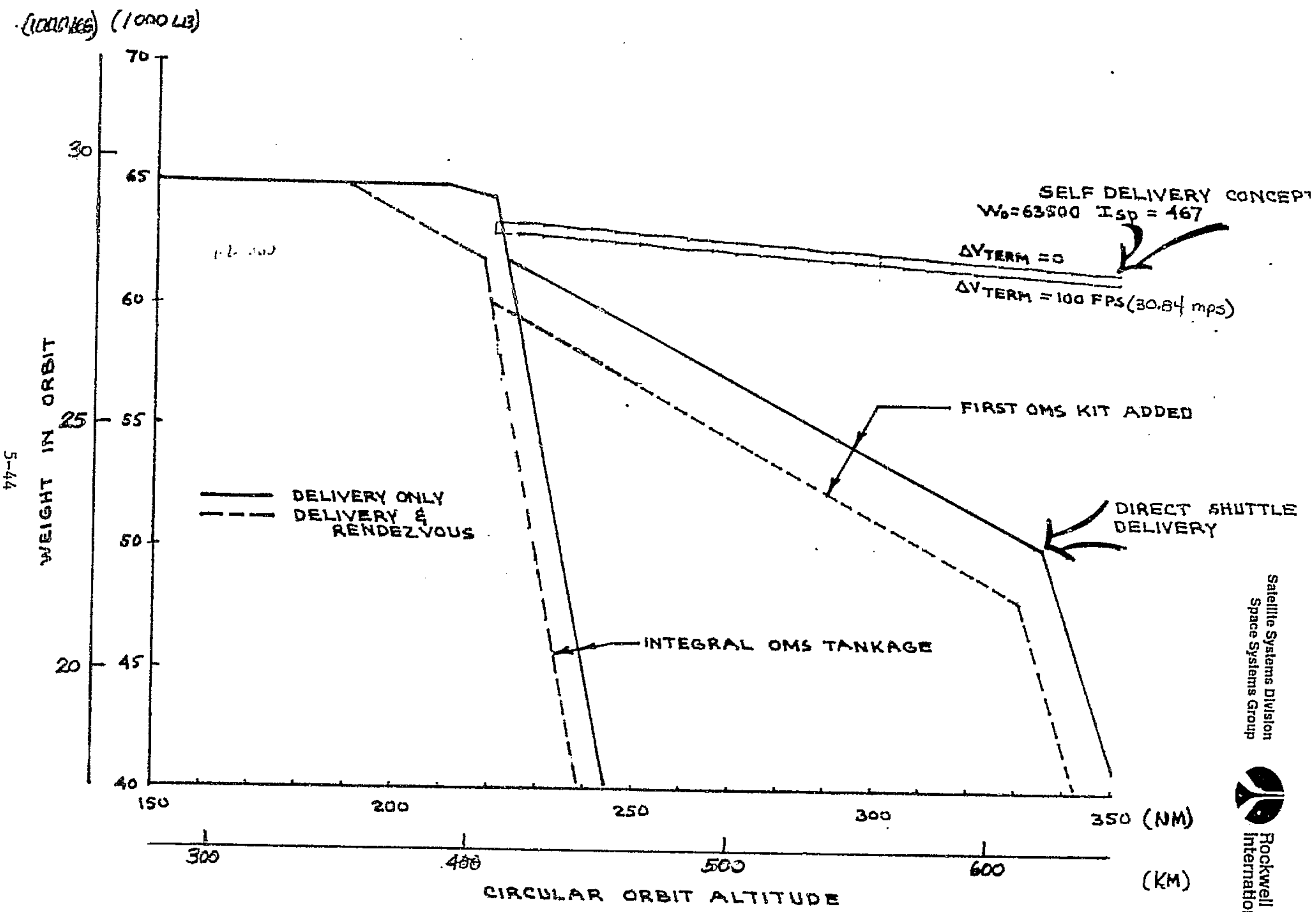
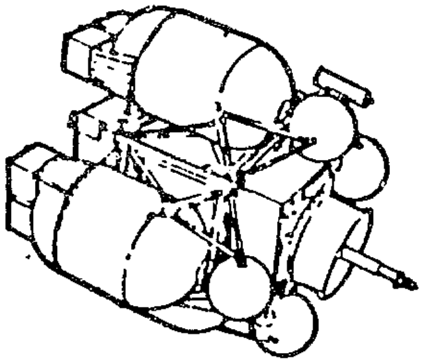


Figure 5.2-14. Cargo Weight in Orbit KSC Launch (Inclination = 28.5 Deg.)



SYSTEM CHARACTERISTICS

- 24 NOZZLE GUIDANCE AND ATTITUDE CONTROL SYSTEM
  - -6 DEGREES OF FREEDOM
- STRAP-ON PROPULSION KITS (4)
- DOCKING PROBE SYSTEM
- COMMUNICATION AND DATA MANAGEMENT
- MANUAL CONTROL CAPABILITY
- RMS GRAPPLING FIXTURE; ASE FITTING
- TV CAMERAS (2); ILLUMINATION SYSTEM
- THERMAL CONTROL SUBSYSTEM

PERFORMANCE DATA

GROSS WEIGHT (WET)	9,900 LBS	4,490 KG
BASIC CORE (WET)	2,300 LBS	1.040 KG
4 BASIC PROPULSION KITS (WET)	7,600 LBS	3,450 KG
DRY WEIGHT	3,440 LBS	1,560 KG
BASIC CORE	1,870 LBS	850 KG
4 PROPULSION KITS	1,570 LBS	710 KG
PROPELLANT: CORE	25,000 LB. SEC.	
(N_2H_4) KITS (4)	1,350,000 LB. SEC.	
PROPULSION KIT THRUST (EACH)	300 LBS	1,330 NEWTONS
RF LINK RANGE	300 MILES	550 KM

$$I_{sp} = \frac{1375000}{6460} = 212.85 \text{ SEC.}$$

- FOR BASELINE OSM + P/L ΔV = 170 TO 190 MPS (570 TO 630 FPS)

Figure 5.2-16. Teleoperator Characteristics

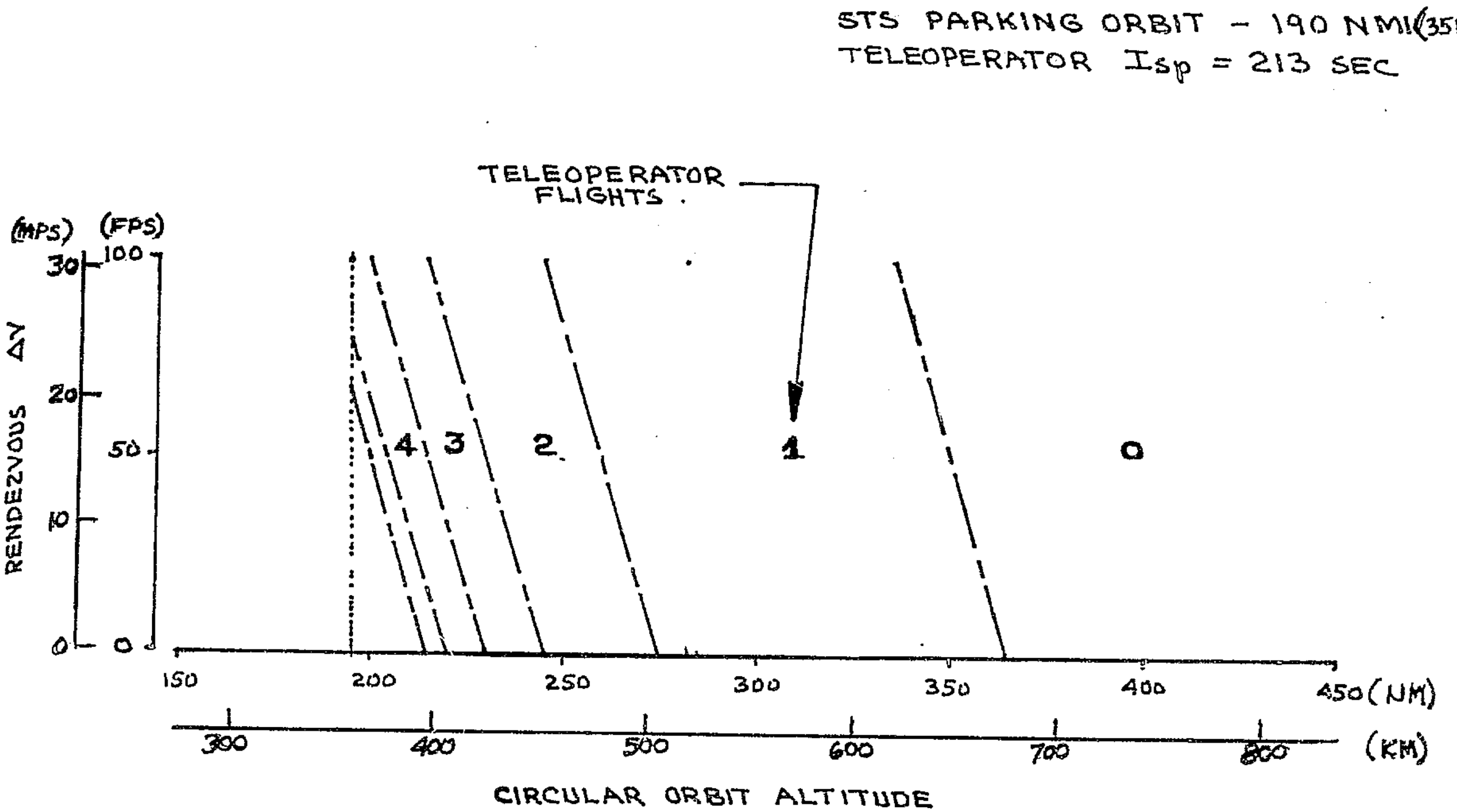


Figure 5.2-17. Number of Teleoperator Flights to Deliver 63,500 lb (28,803 kg) Payload and Return to Parking Orbit

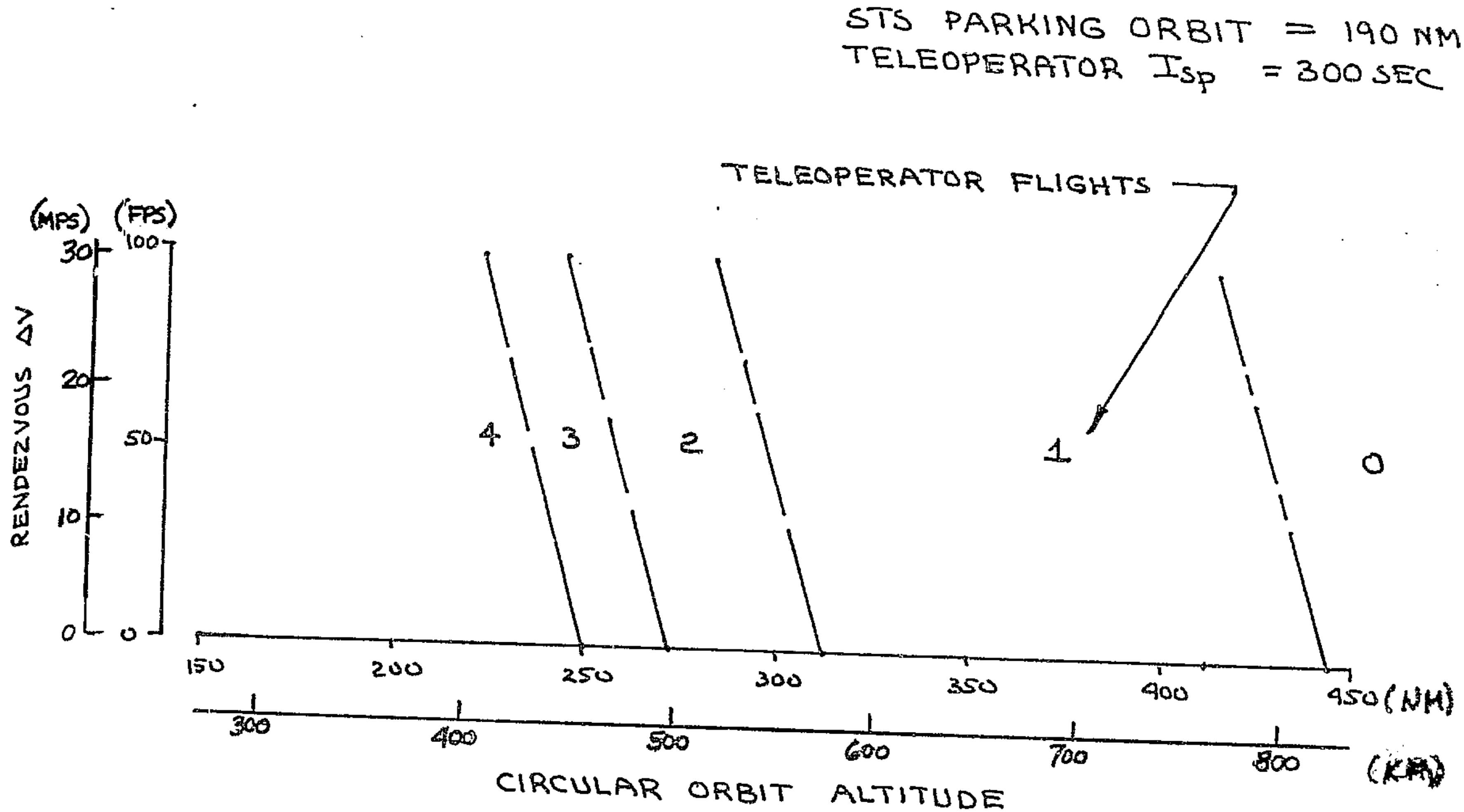


Figure 5.2-18. Number of Teleoperator Flights to Deliver 63,500 lb (28,803 kg) Payload and Return to Parking Orbit

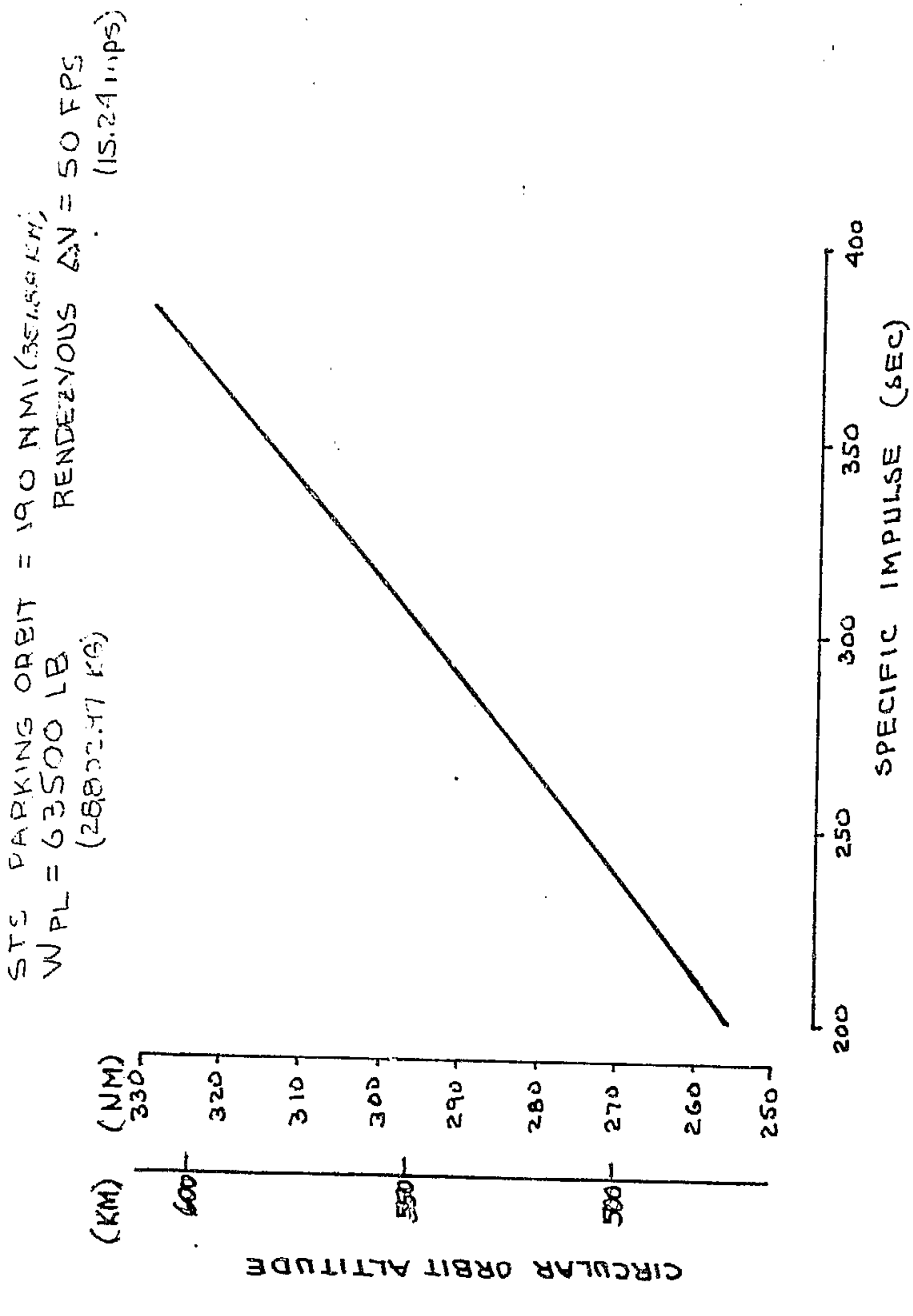


Figure 5.2-19. Teleoperator to Perform Two Delivery Flights

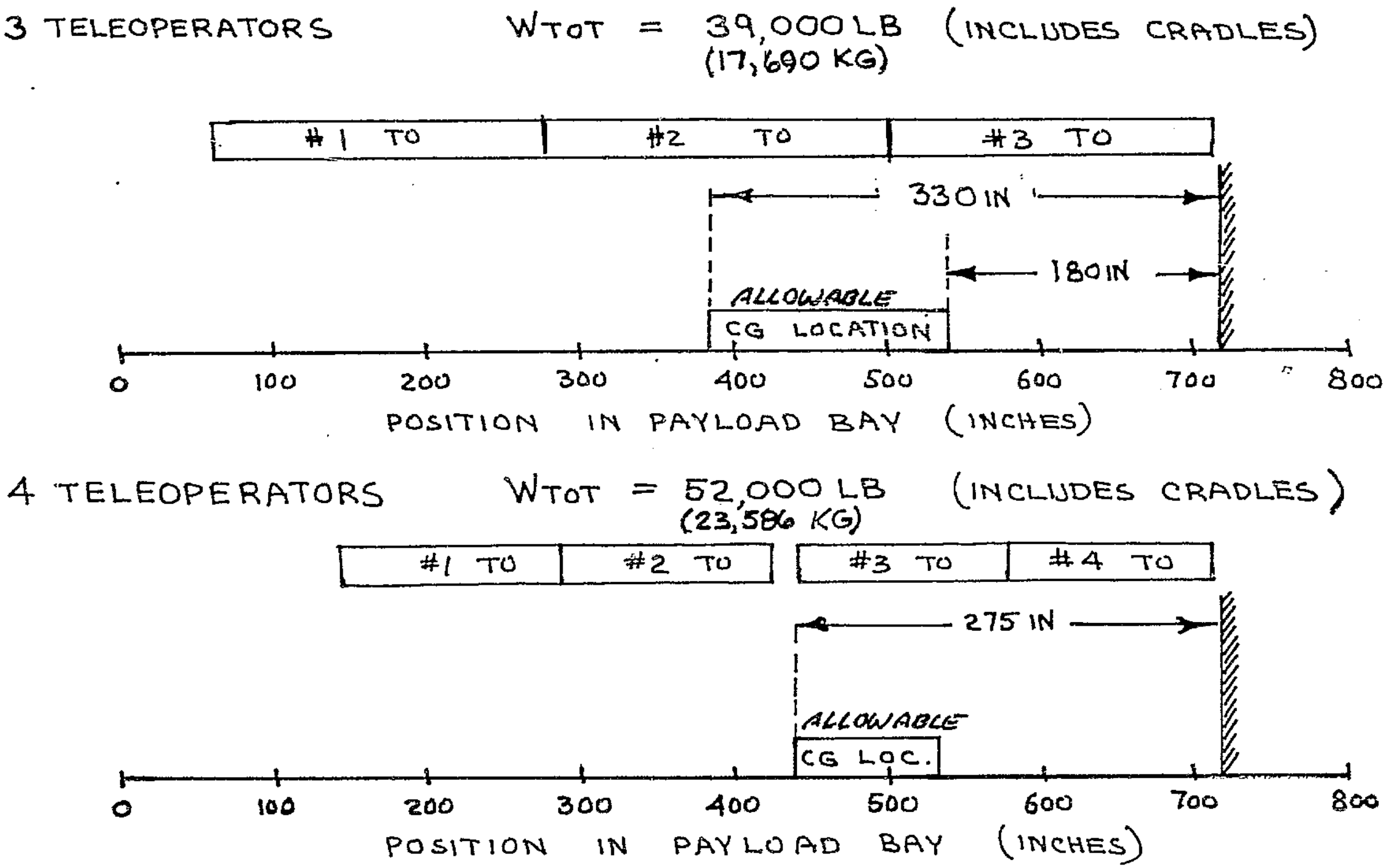


Figure 5.2-20. Teleoperator Placement

teleoperator placement within the bay. Thus, for the four teleoperator combination two teleoperators would have to fit in a 275 inch length to stay within the forword c.g. limit. This would allow 137.5 in. for each teleoperator giving 7.5 in. of end clearance. With only three teleoperators in the bay each individual unit could occupy as much as 220 in. of the payload bay length giving up to 90 in. of end clearance.

- Teleoperator Farm Concept

The delivery of more than one teleoperator by a single Shuttle flight immediately establishes a teleoperator garage or farm ready to take delivery of any subsequent payloads. It is also obvious that unless there is a need to perform a number of delivery missions at the same time, only one "intelligent" core unit is really required. This reduces the total equipment weight that is needed at any one time in orbit.

Thus, the operation of a teleoperator farm would differ from individual teleoperator operation in that only a single "intelligent" core vehicle would be used. The required propellant may be either space stored in a single "gas station" craft that would refuel the teleoperator vehicle after each delivery mission. Or as an alternate technique the propellant could already be stored in plug-in replacement tanks for the core vehicle. In this later method the used empty propellant tanks could be either discarded or returned to Earth to be refilled and reused. The plug-in tank concept seems to offer a lower technology risk than the propellant transfer in orbit.

The farm would have to remain in orbit for the duration of the delivery phase of the mission (the delivery of five propulsion modules). The location of such a farm would primarily be driven by the relative precession of the two orbits of interest, i.e., the farm orbit and the platform construction orbit. Although the orbits would be at the same inclination, their different altitudes will cause differences in nodal precession rates thereby leading to an out-of-plane or relative inclination condition. An example relative inclination history is shown in Figure 5.2-21. The orbits are both at 28.5 deg. inclination, but their altitudes are 370 km (200 nmi) and 610 km (330 nmi) respectively. The rate of change of the relative inclination angle between the two orbits is approximately 0.38 degrees per day for the first 100 days. The relative inclination rate between the construction orbit and potential teleoperator farm orbits is given as function of the farm orbit altitude in Figure 5.2-22. These rates are only valid when relative inclination is less than 30 degrees. As can be seen in Figure 5.2-21 eventually the two orbits will again be coplanar, but the time period when this would reoccur is much longer than the postulated mission period.

The increase in transfer velocity for relative inclinations of less than 4 degrees is shown in Figure 5.2-23. Although the inclination difference is small the transfer velocity increase is considerable, approximately doubling for Δi of less than 2 degrees.

In order to have the relative inclination between the farm and construction orbit to remain essentially fixed, i.e., the orbits to remain coplanar, the teleoperator farm must be located at the same altitude as the platform, possibly even berthed to the platform. The teleoperators then would always be

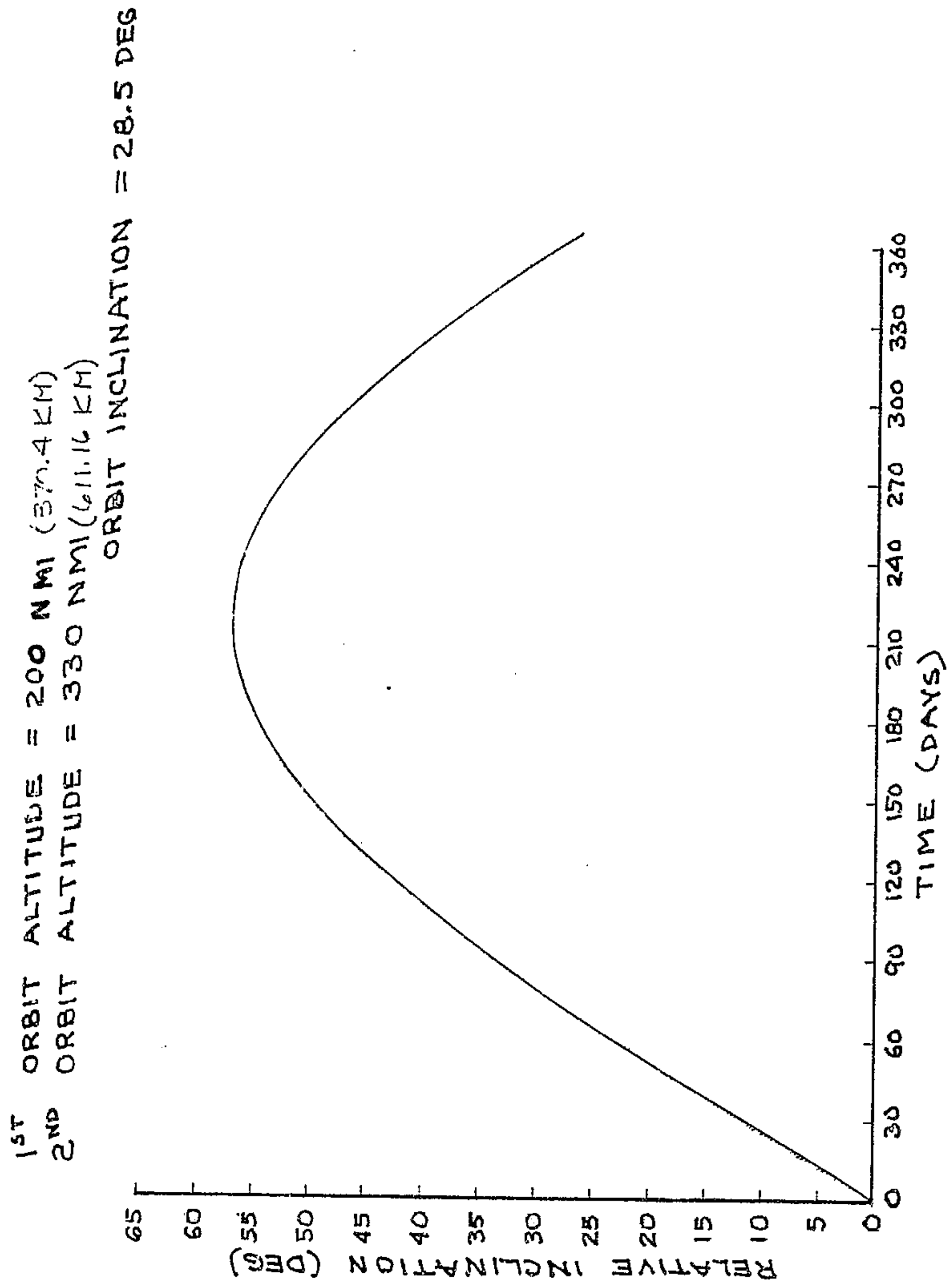


Figure 5.2-21. Relative Inclination

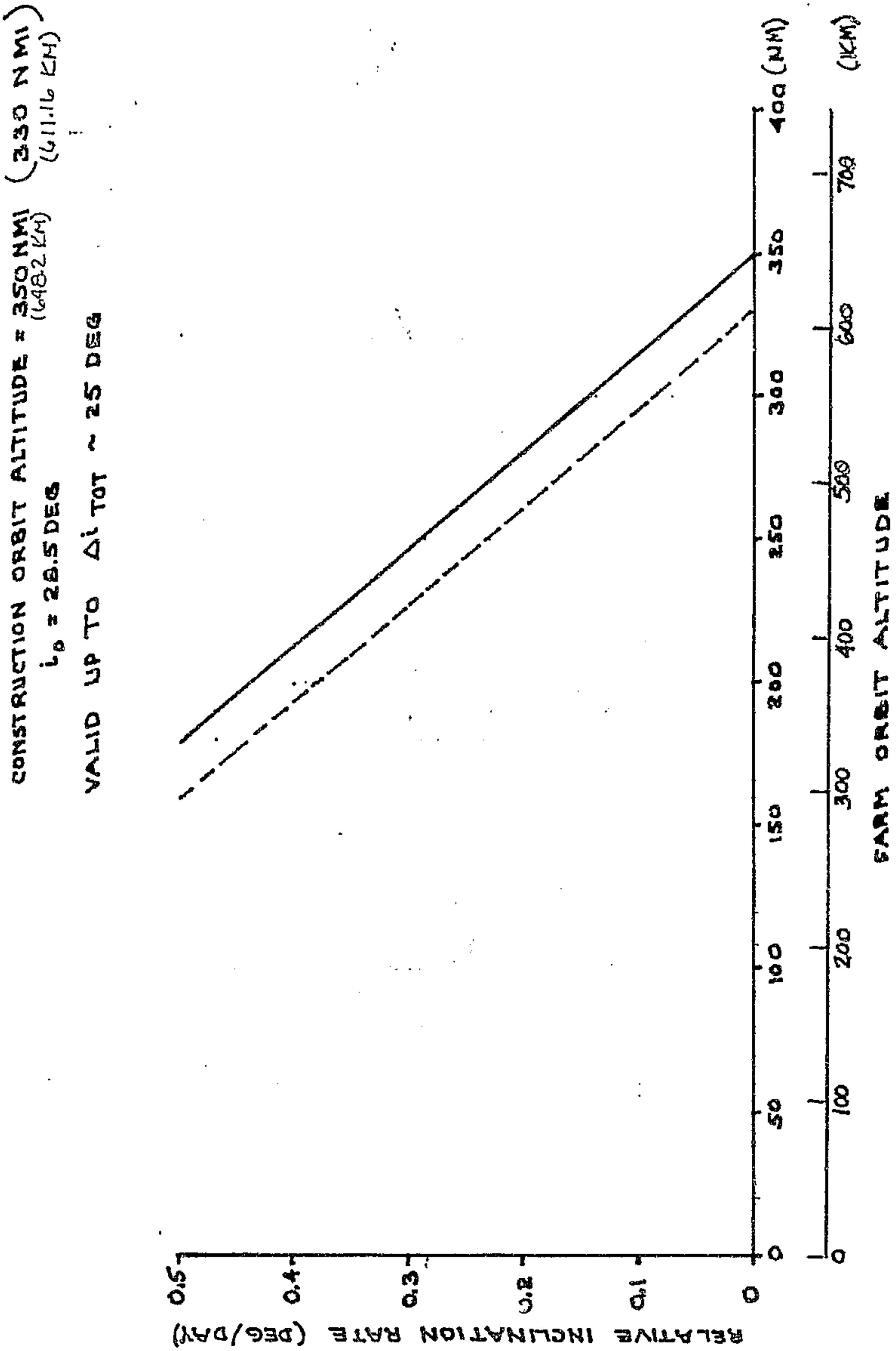


Figure 5.2-22. Relative Inclination Rate

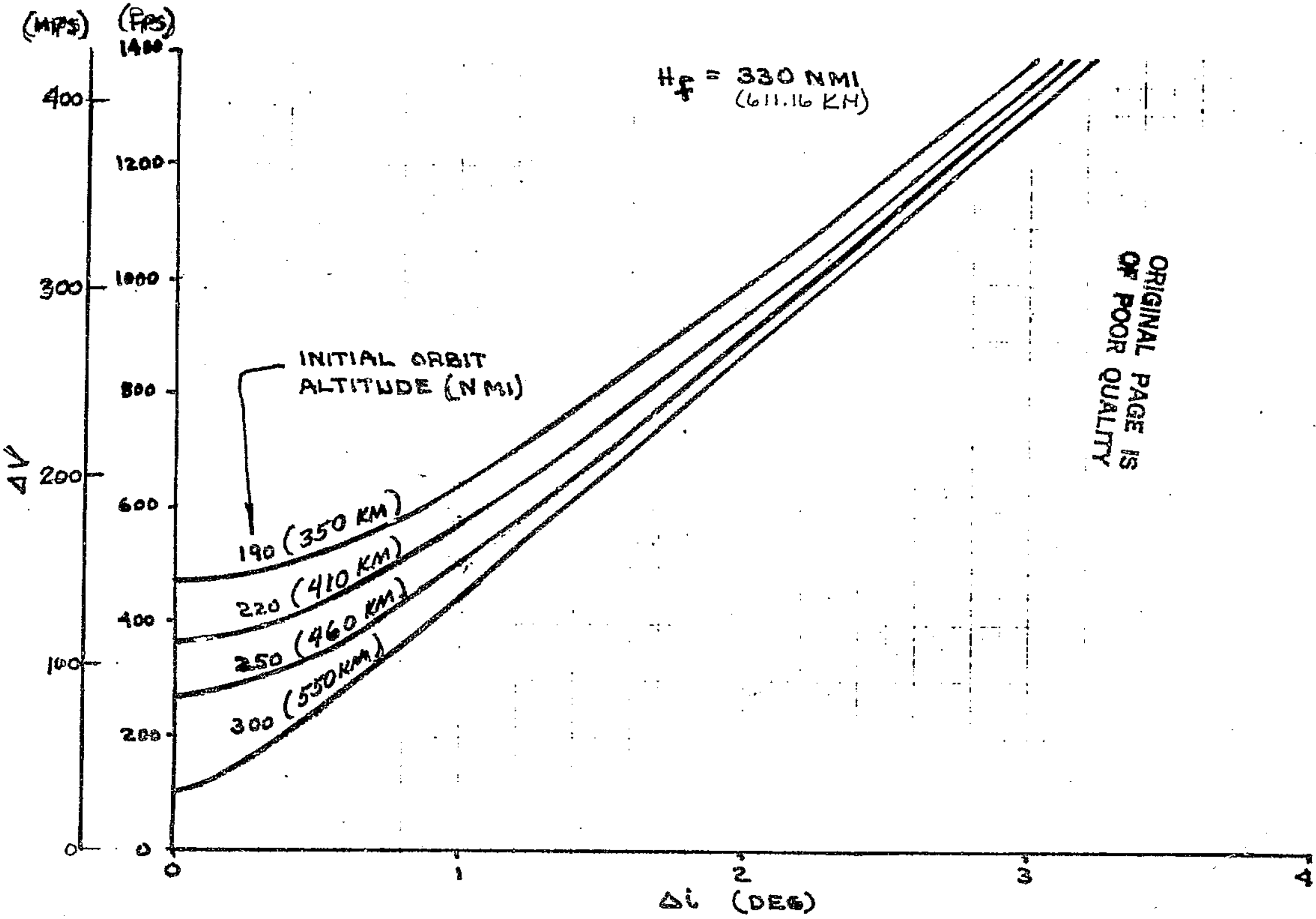


Figure 5.2-23. Transfer Velocity Requirement
(No Rendezvous ΔV Included)

coplanar with the platform and it would be up to the Shuttle launch control to guarantee a nearly in plane rendezvous mission. This would be achieved by proper choice of the Shuttle launch time.

A pictorial representation of the teleoperator farm concept mission profile is shown in Figure 5.2-24. A fully loaded teleoperator at the construction site altitude would descend and rendezvous with an Orbiter at 350 km (190 nmi) that has just brought up one propulsion module. The propulsion module is deployed by the Orbiter and handed over to the awaiting teleoperator. The teleoperator then delivers and berths the fully loaded propulsion module to the platform. The nearly spent teleoperator then completes the mission cycle by returning to the farm. It is then refueled and remains there awaiting the next Orbiter flight.

The farm size that is required to deliver five fully loaded propulsion modules to the construction site is given in Figure 5.2-25 as a function on the teleoperator specific impulse. The structural efficiency of the propellant tanks and the weight of a single "intelligent" core unit was assumed to be the same as for the Martin teleoperator concept. Even for the lowest energy teleoperator (the Martin concept) which has a specific impulse of 213 sec the entire teleoperator farm would weigh less than 18,100 kg (40,000 lb). A single Shuttle flight could deliver this low energy teleoperator farm to the construction site altitude (610 km, assumed) and still allow up to 3630 kg (8000 lb) for any necessary auxiliary equipment and/or cradle. Higher energy teleoperator farms, weighing considerably less, could be delivered with much greater ease.

- Single Superteleoperator Delivery Concept

Instead of delivering each individual propulsion module to the constructed platform as it is launched by the Shuttle some advantage may be gained by first delivering all the propulsion modules to a low orbit and then boosting them as a unit to the higher construction orbit. Timing and careful mission planning again dominate this concept.

The fact that the propulsion modules are assembled at a lower orbit altitude than the platform again introduces nodal regression into the problem. A change in relative inclination will occur as a function of time. The boost to the construction orbit after all the modules have been assembled must occur when the two orbits are nearly coplanar. Otherwise larger ΔV requirements will result in greater propellant weights.

Unplanned, unexpected lengthy delays in the placement of the propulsion modules in the lower orbit would result in the delay of the transfer past the predetermined time. Such delays of more than a few days could not be tolerated.

The weight of a superteleoperator that would perform the transfer mission is shown in Figure 5.2-26. Both a one way mission where the teleoperator is left with the platform and a two way mission where the teleoperator is returned for subsequent reuse are shown. The one way mission ΔV, that includes 15 mps (50 fps) for rendezvous and docking comes to 159 mps (523 fps).

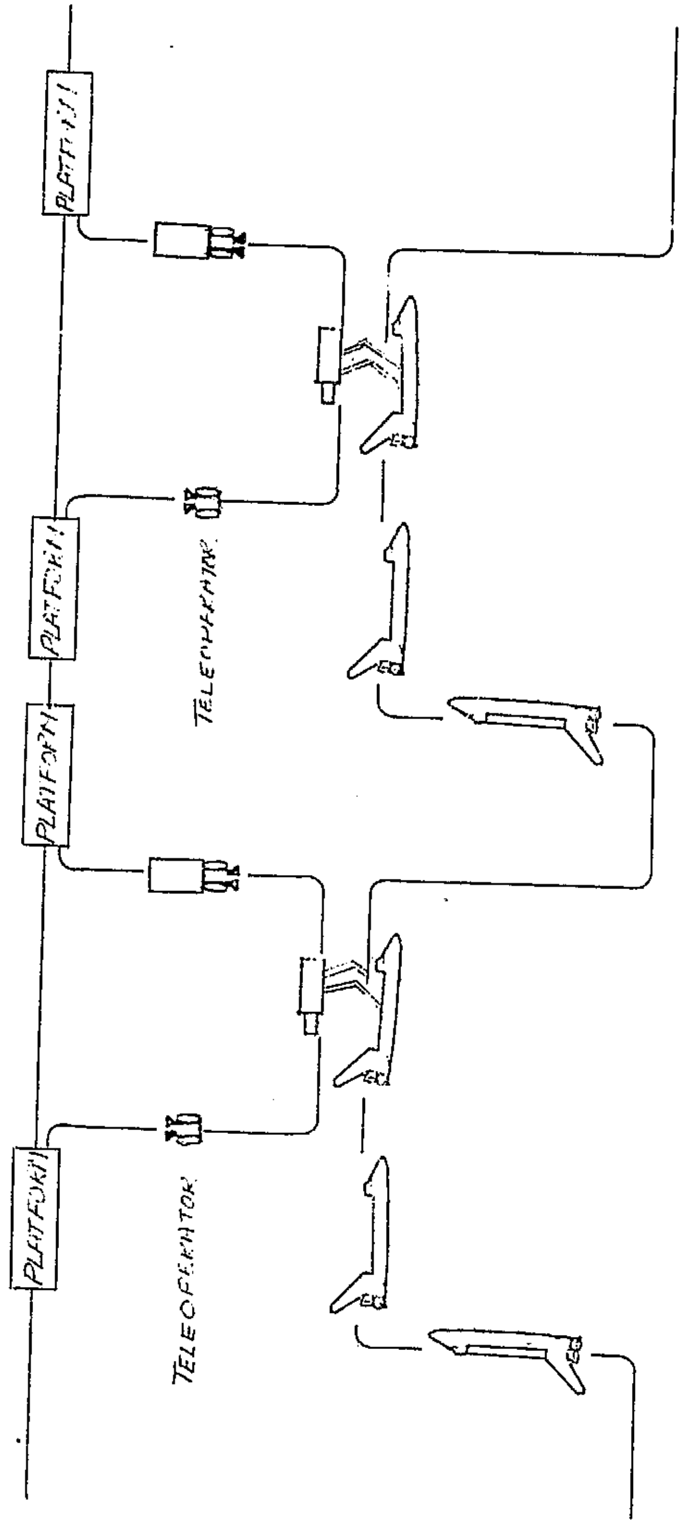


Figure 5.2-24. Teleoperator Farm Concept Mission Profile

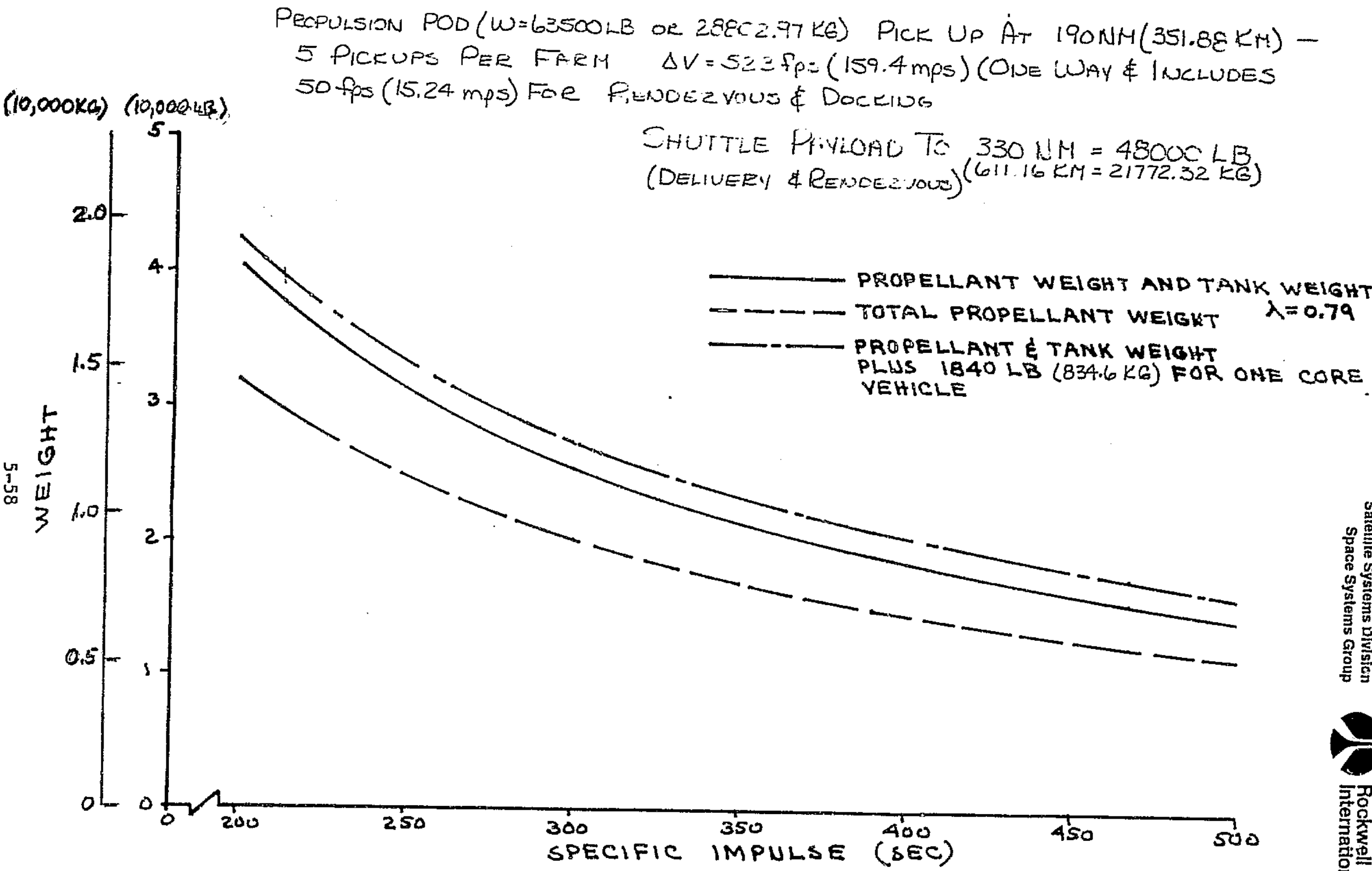


Figure 5.2-25. Teleoperator Farm at 330 nmi (611.16 km)

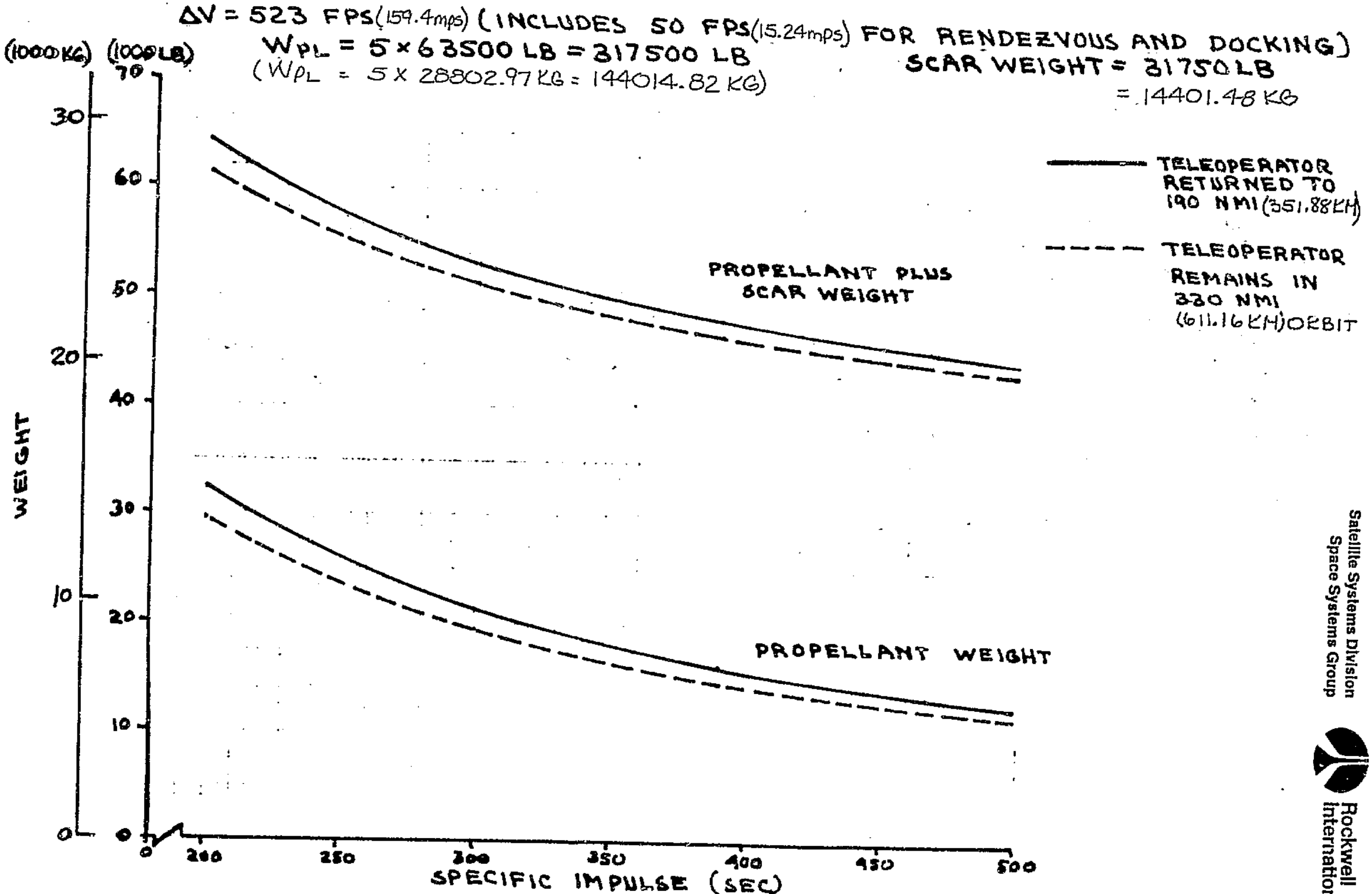


Figure 5.2-26. Super Teleoperator Delivery From 190 nmi (351.88 km) to 330 nmi (611.16 km)

The total superteleoperator weight includes a conservative estimate of the teleoperator structure that would be required to keep all the propulsion modules together. This was estimated to be 10% of the total propulsion system weight or 14,400 kg (31,750 lb). This weight also approximately corresponds to the maximum returnable weight for the Orbiter. With this conservatism built into the system the total superteleoperator weight would be only 28,100 kg (62,000 lb), allowing the unit to be placed into orbit by a single Shuttle.

• Platform "Fly Down"

In all the previously discussed concepts the propulsion modules (144,000 kg or 317,500 lb) were to be delivered to the construction site of the completed platform. This last concept reverses the technique. The completed platform which weighs 60,500 kg (133,400 lb) is to be transferred to the lower Shuttle delivery compatible orbit. The teleoperator weight to accomplish this task is shown in Figure 5.2-27. For the low energy teleoperator (specific impulse of 213 sec) the required weight is only 4990 kg (11,000 lb), smallest of all the concepts.

Positive transfer control would be achieved by using the platform guidance unit. Rendezvous and docking would be performed by the Orbiter.

Two critical aspects for this delivery mode still need to be evaluated. The first of these is the effect of atmospheric drag on the payload platform and the subsequent requirements of orbit altitude maintenance while the five propulsion modules are being delivered by the Orbiter.

Representative orbit decay effect is shown in Figure 5.2-28. For various ballistic coefficients (W/C_DA) the time the altitude would decay 10% is shown as function of orbit altitude. Five Orbiter flights would take approximately eight weeks.

If orbit-keeping, i.e., orbit altitude maintenance is required during this period, additional propellant will be necessary. The propellant required to maintain altitude is shown in Figure 5.2-29. Data are shown for ballistic coefficients of 1 and 25 psf and are based on a mission duration of 8 weeks and a specific impulse of 300 sec. The addition of these propulsion modules during the delivery porcess will tend to increase the ballistic coefficient. The average value for the platform configuration studied here is approximately 27 psf.

The second potential area of concern is the drag effect during the low thrust transfer to geosynchronous orbit. The lower altitude would result in further degradation of the already low thrust-to-weight ratio. However, this too could be overcome by additional propellant.

Delivery Concept Summary

The most promissing of the delivery concepts and some of the sensitivities with respect to propellant type (specific impulse) have been discussed in the preceding sections.

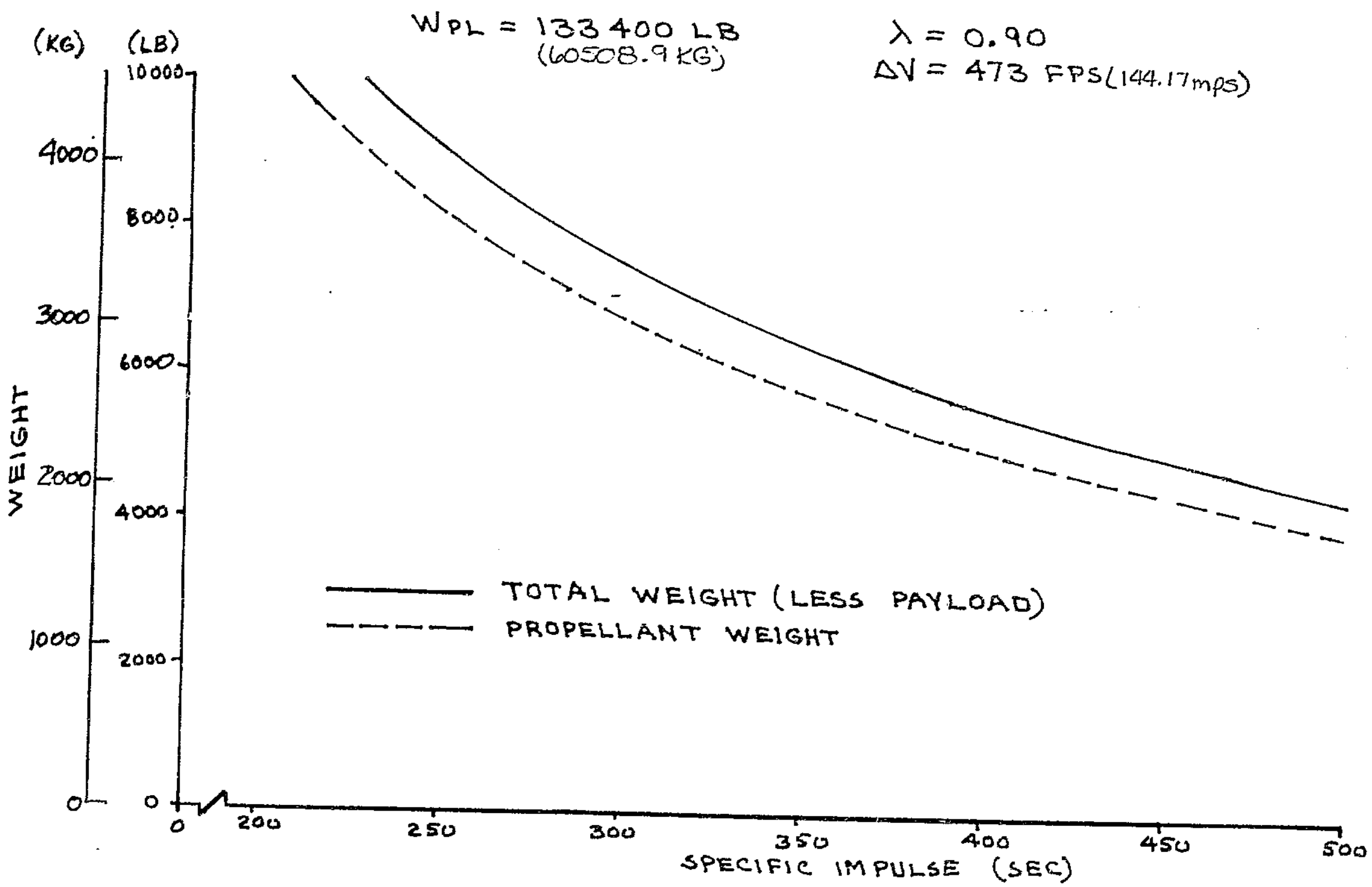


Figure 5.2-27. Platform Transfer From 330 nmi (611.16 km) to 190 nmi (351.88 km)

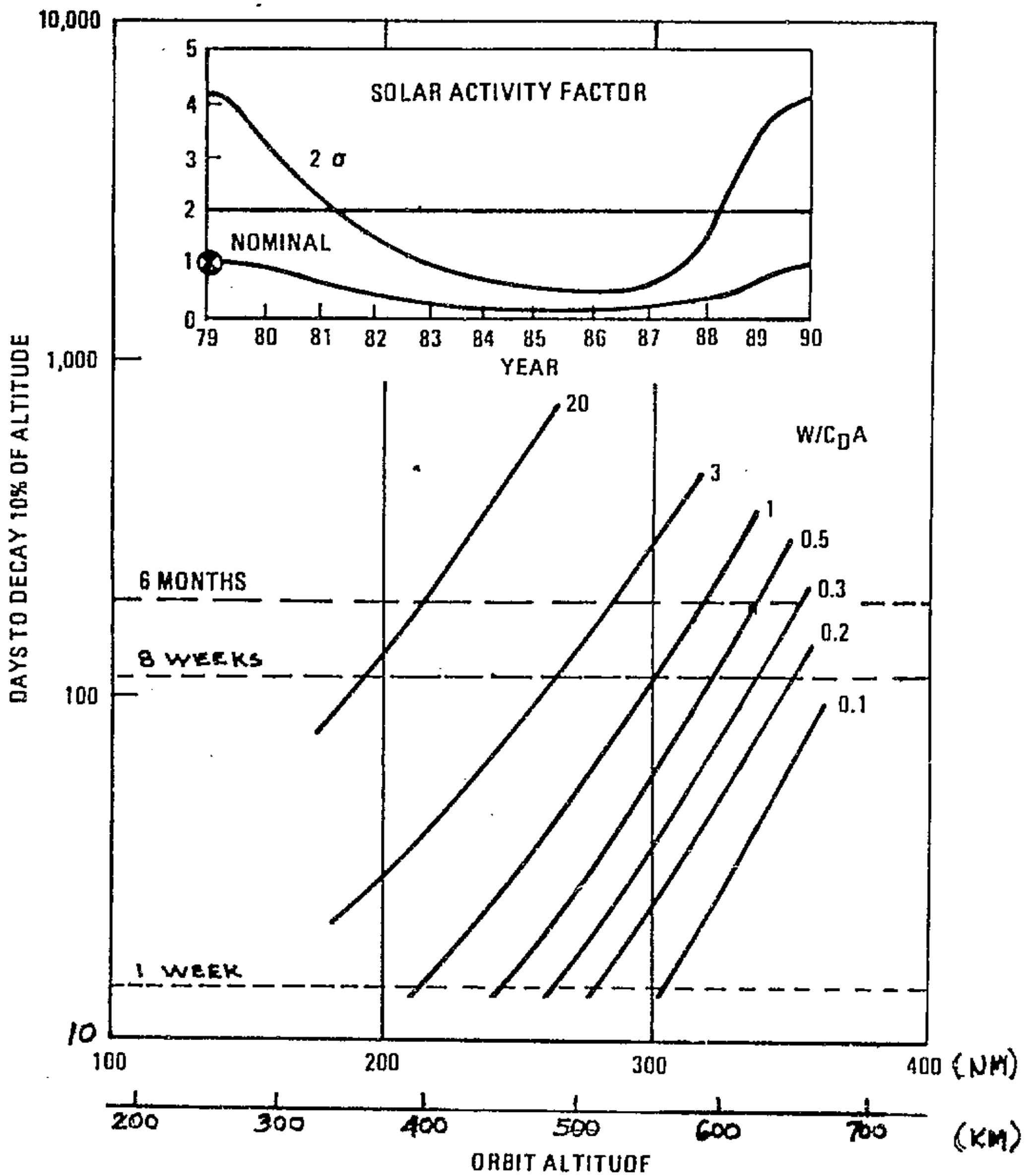


Figure 5.2-28. Orbit Lifetime

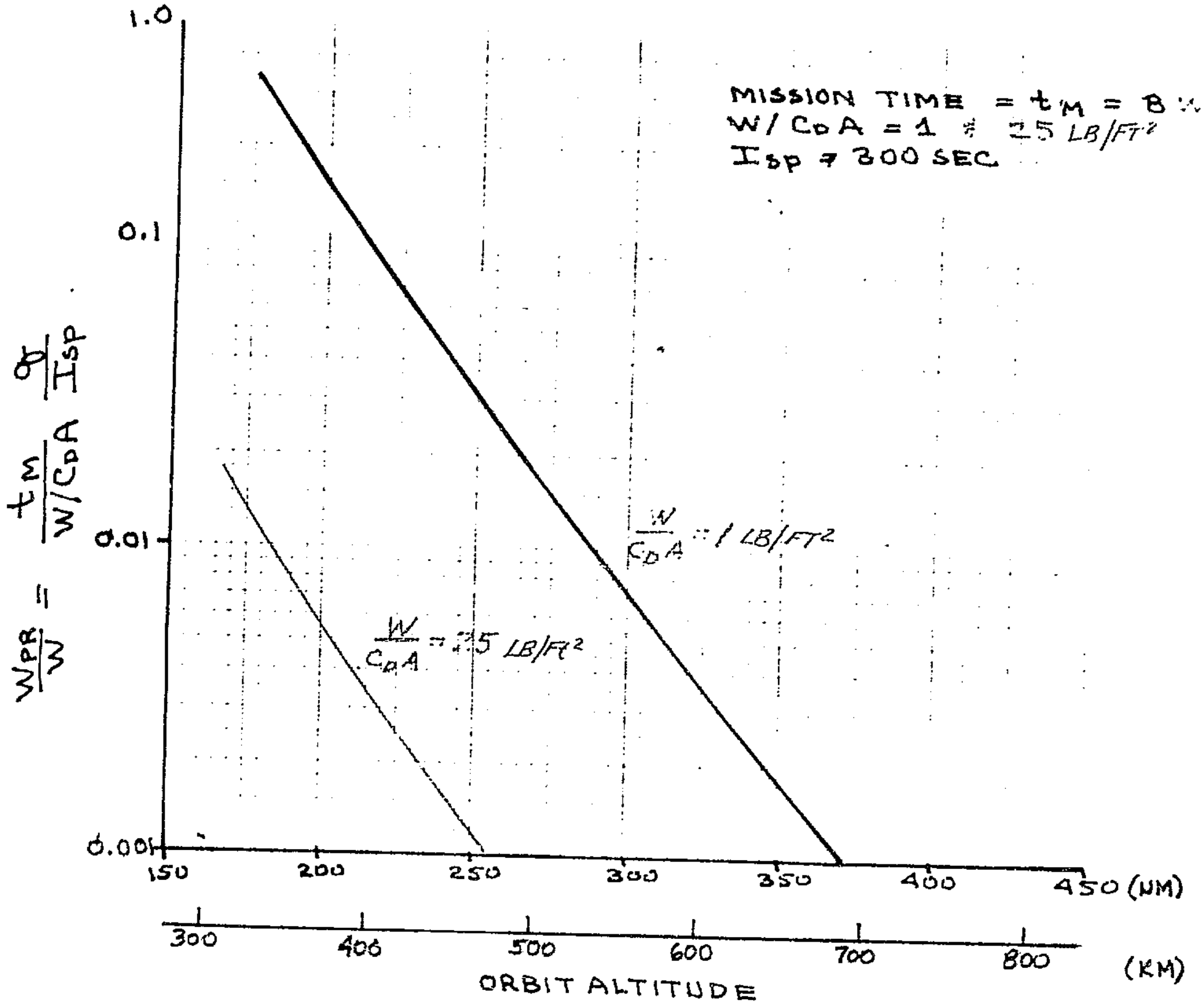


Figure 5.2-29. Orbit Altitude Maintenance Propellant
Nominal Maximum Solar Activity

Satellite Systems Division
Space Systems Group

Rockwell International

A summary showing the total number of Shuttle flights that would be required to deliver a total of 5×28,800 = 144,000 kg (5×63,500 = 317,500 lb) of propulsion modules to the constructed platform has been prepared in Table 5.2-3. In this summary only low technology, monopropellant teleoperator concepts are included.

The Direct Shuttle Delivery Concept and the Individual Teleoperator Delivery Concept each require seven Shuttle flights. There exists, however, a possibility that for a higher energy individual teleoperator delivery concept one of the Shuttle flights could be eliminated.

All other concepts require six Shuttle flights. With the exception of the Self Delivery mode where six individual propulsion modules are to be delivered, the remaining concepts require five Shuttle flights to deliver the five propulsion modules to orbit and then one additional flight to deliver the teleoperator hardware. Improvements in specific impulse will not change this ratio.

In the platform fly-down concept the teleoperator weight represents only 23% of the Shuttle capability to 610 km (330 nmi) altitude. It is possible therefore that the teleoperator delivery could be combined with one of the construction materials delivery flights.

The platform fly-down concept could also be expanded to consider an approach in which the platform RCS subsystem could be enlarged sufficiently to allow the platform to fly itself down to the Shuttle delivery orbit. Since the platform must have orbital maneuvering capability in order to perform stationkeeping as part of its GEO mission, it would already have the functional ability to perform this maneuver. This platform self fly-down concept would likely be the most efficient of all the identified techniques in terms of total weight to orbit. Also, it is likely that the increased weight of the RCS modules would have little impact on the construction delivery operations since most construction missions tend to be volume limited. In addition to its likely high logistics efficiency the platform self-fly-down concept would also offer the man in-situ advantages of the direct shuttle delivery approach. Thus, the platform fly-down concept appears to be a leading candidate for serious consideration of propulsion delivery methods.

5.2.1.8 References

(1) Space Construction Systems Analysis Project Systems Review, Rockwell International Corporation, Satellite Systems Division, PD 79-08, March 1979.

(2) Orbit-to-Orbit Shuttle (Chemical) Feasibility Study, Final Report, Vol. IV—System Design, Part 2—System Design Analysis, SAMSO-TR-71-238. Vol. IV-2, North American Rockwell, Space Division (October 1971).

(3) DOD Upper Stage/Shuttle System Preliminary Requirements Study, Briefing Manual, SAMSO-TR-72-202, McDonnell Douglas Astronautics Co. (Aug. 1972).

(4) Baseline Tug Definition Document, George C. Marshall Space Flight Center, 1974.

(5) OTV, Large Space Structures and Other Space Applications, Propulsion, Rockwell International Corporation, Rocketdyne Division, BC 78-92, June 1978.

Table 5.2-3. Alternate Delivery Concept Summary

DELIVERY CONCEPT	NUMBER SHUTTLE FLIGHTS	PAYLOAD WT. KG (LB)	AUX. EQUIP. WT. KG (LB)	ALT KM (NMI)
DIRECT SHUTTLE DELIVERY	7	20,600 (45,400)	1,180 (2,600)	610 (330)
SELF DELIVERY	6	24,000 (53,000)	5,440 (12,000) [1]	407 (220)
INDIVIDUAL TELEOPERATORS	5	28,800 (63,500)	680 (1,500)	350 (190)
(THREE DELIVERED)	1	13,500 (29,700)	4,200 (9,300)	350 (190) [2]
(TWO DELIVERED)	1	9,000 (19,800)	2,812 (6,200)	350 (190) [2]
TELEOPERATOR FARM	5	28,800 (63,500)	680 (1,500)	350 (190)
	1	18,100 (40,000)	3,630 (8,000	610 (330)
SUPERTELEOPERATOR	5	28,800 (63,500)	680 (1,500)	350 (190)
	1	28,100 (62,000)	1,360 (3,000)	350 (190)
PLATFORM FLY-DOWN	5	28,800 (63,500)	680 (1,500)	350 (190)
	1	4,990 (11,000) [3]	16,800 (37,000) [3]	610 (330)

NOTES:
(1) INCLUDES THE "INTELLIGENT UNIT REQUIRED FOR PROP. MODLS.
(2) IF THIS CONCEPT IS SELECTED, TELEOPERATOR ALTITUDE WILL HAVE TO BE REEVALUATED.
(3) TELEOPERATOR IS ONLY 23% OF SHUTTLE CAPABILITY.

5.2.2 Advanced Storables

The applicability of other than cryogenic propellants has been analyzed for the large structure platform delivery from the low earth construction orbit to the geosynchronous equatorial mission orbit. In particular, the use of space storable liquid propellants was evaluated.

5.2.2.1 Technology Considerations

The cryogenic propellants, LO_2-LH_2, discussed in the previous sections, represent the high end of the energy spectrum for chemical propulsion. The cryogenic propellants, however, besides having a very low bulk density, are also difficult to store for long periods of time because of the inherent extremely low storage temperature requirements (<-300°F). Propellants that remain in liquid form at higher temperatures (>+60°F) are considered space storable. Most common in this class of propellants is the combination of nitrogen-tetroxide as the oxidizer and one of the hydrazines as the fuel. There is considerable experience being developed in handling these propellants. For example, the Space Shuttle OMS engines use nitrogen-tetroxide and monomethylhydrazine as propellants. The specific impulse for the OMS is 313 sec. The theoretical maximum for this propellant combination is approximately 340 sec. In general, as illustrated in Figure 5.2-30, the storable propellants have a lower energy content than the cryogenics, but they do have a much higher bulk density. High bulk density will result in lower tank volume for a given propellant weight and, thus, will represent smaller and ligher tank structure (i.e., higher structural efficiency). Structural efficiency factors based on projection from existing hardware techniques and design considerations are illustrated in Figure 5.2-31. Based on typical bulk densities for storable propellants (Figure 5.2-30), theoretically feasible structural efficiency factors range between 0.92 and 0.94. However, additional structural penalties may be incurred if the vehicle is expected to perform additional or unusual tasks. These conditions can generally be accounted for by a structural efficiency factor decrease of 0.01 to 0.02.

In practice, the more conservative approach to structural efficiency seems prudent. For this reason it was assumed that only 0.88 to 0.92 structural efficiency will be achieved for advanced storable propulsion concepts. Similarly, the reasonably achievable specific impulse range was taken to be 313 to 340 sec.

The effect of one propulsion system parameter, burn time, as affected by mission-related parameters, thrust-to-weight ratio, and required mission velocity is illustrated in Figure 5.2-32. Missions with lower average thrust-to-weight ratio will require thrusters with longer operating times. NASA Lewis Research Center is currently sponsoring studies on low-thrust, long-duration engines using cryogenic propellants. Engine run times are shown in Figure 5.2-33 for the thrust range of interest in the study. Engine characteristics using storable propellants probably would be expected to fall in the same general range. For example, the Shuttle OMS engine [26,700-N (6000-lb) thrust] is designed for 30 minutes continuous operation, while in Figure 5.2-33, the 22,200-N (5000-lb) engine is expected to operate continuously for one hour.

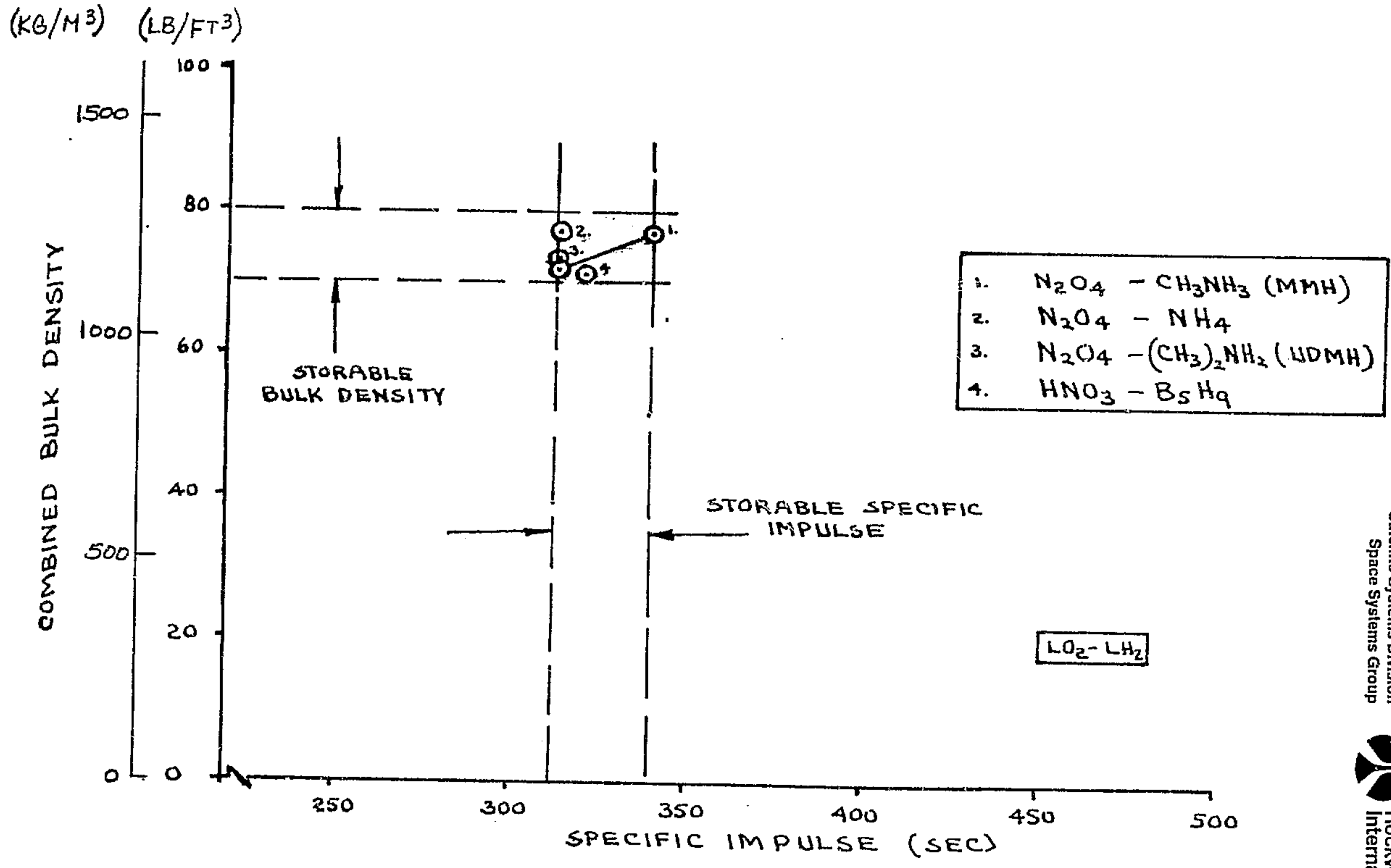


Figure 5.2-30. Storable Propellants

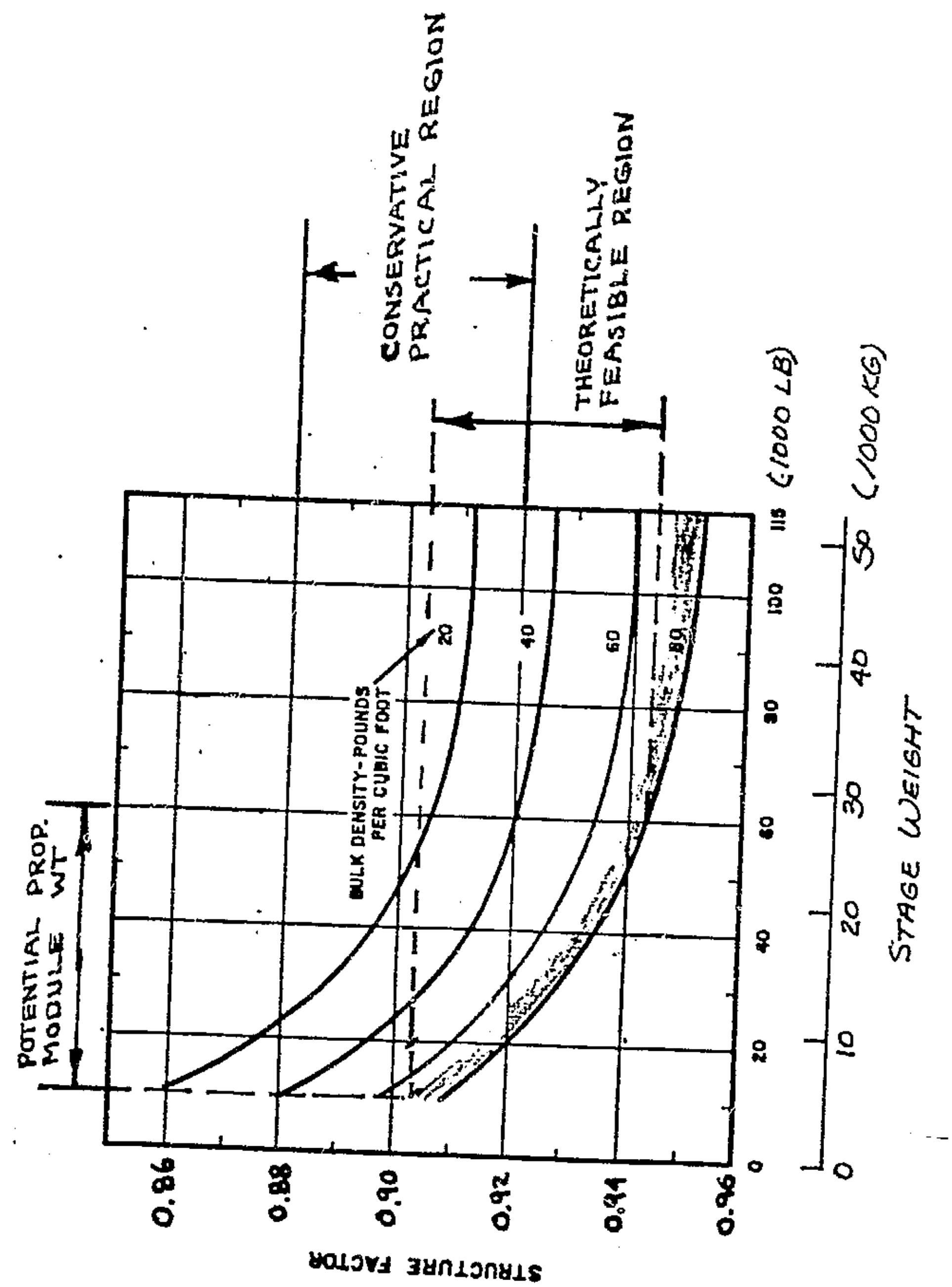
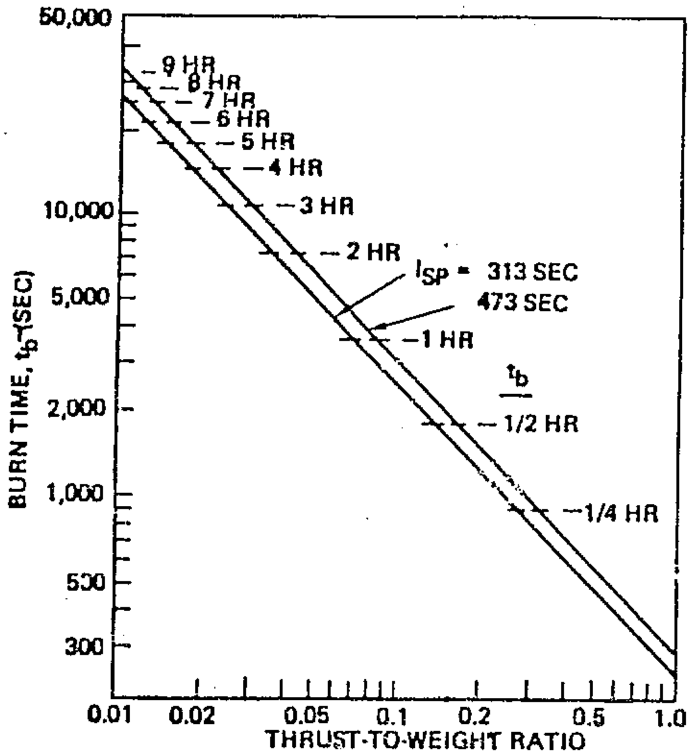


Figure 5.2-31. Structure Factor



T/W	*ΔV (FT/SEC)
0.01	18,250
0.1	14,650
1.0	14,000

*ONE-WAY TRIP, LEO-TO-GEO

Figure 5.2-32. Burn Time

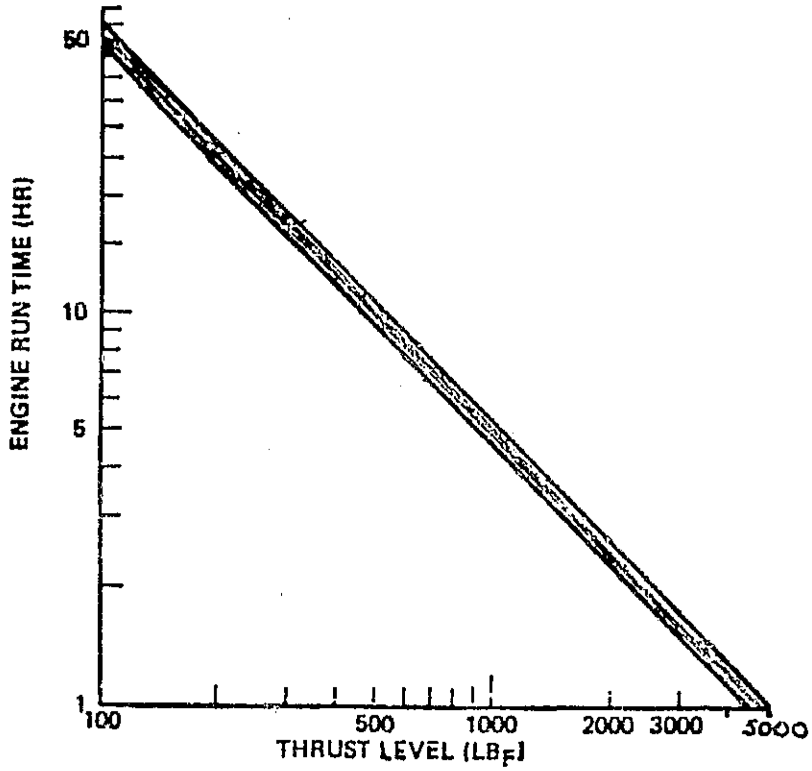


Figure 5.2-33. Accumulative Engine Run Time

It should be noted that extremely long burn times can be expected to be achieved only by the lower thrust engines. Thus, to match engine operation time with the required burn time to achieve the mission objectives, clustering of low-thrust engines to increase the average thrust-to-weight ratio or tandem operation of higher thrust engines with lower engine operation times, might have to be considered.

5.2.2.2 Propulsion Module Sizing

The mission velocity requirement is the unique mission parameter that dominates the overall sizing analysis. The impulsive transfer velocity requirements, representative of thrust-to-weight ratios equal to or greater than one, are shown in Table 5.2-4. For transfer from a low earth orbit (≈500 km) to a geosynchronous orbit with a 28.5-degree plane change included in the maneuver, the smallest mission velocity required is approximately 4120-4270 mps (13,500-14,000 fps), depending on the actual details of the mission contemplated.

The effect of lower average thrust-to-weight ratio is to increase this requirement as illustrated in Figure 5.2-34. For example, for an average thrust-to-weight ratio of 0.001 this increase is nearly 1200 mps (4000 fps). For even lower thrust-to-weight ratios, representing primarily electric propulsion systems with continuous thrust during the transfer, the additional velocity requirement can be as high as 1740 mps (5700 fps) [ΔV_{total} = 6000 mps (19,700 fps)].

Table 5.2-4. Impulsive ΔV Requirements for LEO-to-GEO Transfers

	No Plane Split		Optimum Plane Split	
Parking orbit (circular)				
Altitude, km (nmi)	351.88 (190)	611.16 (330)	351.88 (190)	611.16 (330)
Inclination (degrees)	28.5	28.5	28.5	28.5
Transfer orbit (elliptical)				
Perigee altitude, km (nmi)	351.88 (190)	611.16 (330)	351.88 (190)	611.16 (330)
Apogee altitude, km (nmi)	35,786.2 (19,323)	35,786.2 (19,323)	35,786.2 (19,323)	35,786.2 (19,323)
Inclination (degrees)	28.5	28.5	26.3	26.3
Final orbit (circular)				
Altitude, km (nmi)	35,786.2 (19,323)	35,786.2 (19,323)	35,786.2 (19,323)	35,786.2 (19,323)
Inclination (degrees)	0	0	0	0
ΔV_1 (parking orbit to transfer orbit) mps (fps)	2411.58 (7912)	2340.25 (7678)	2435.32 (7990)	2363.72 (7755)
ΔV_2 (transfer orbit to final orbit) mps (fps)	1827.28 (5995)	1811.43 (5943)	1778.51 (5835)	1761.74 (5780)
ΔV_3 Total, mps (fps)	4238.85 (13,907)	4151.68 (13,621)	4213.86 (13,825)	4125.47 (13,535)

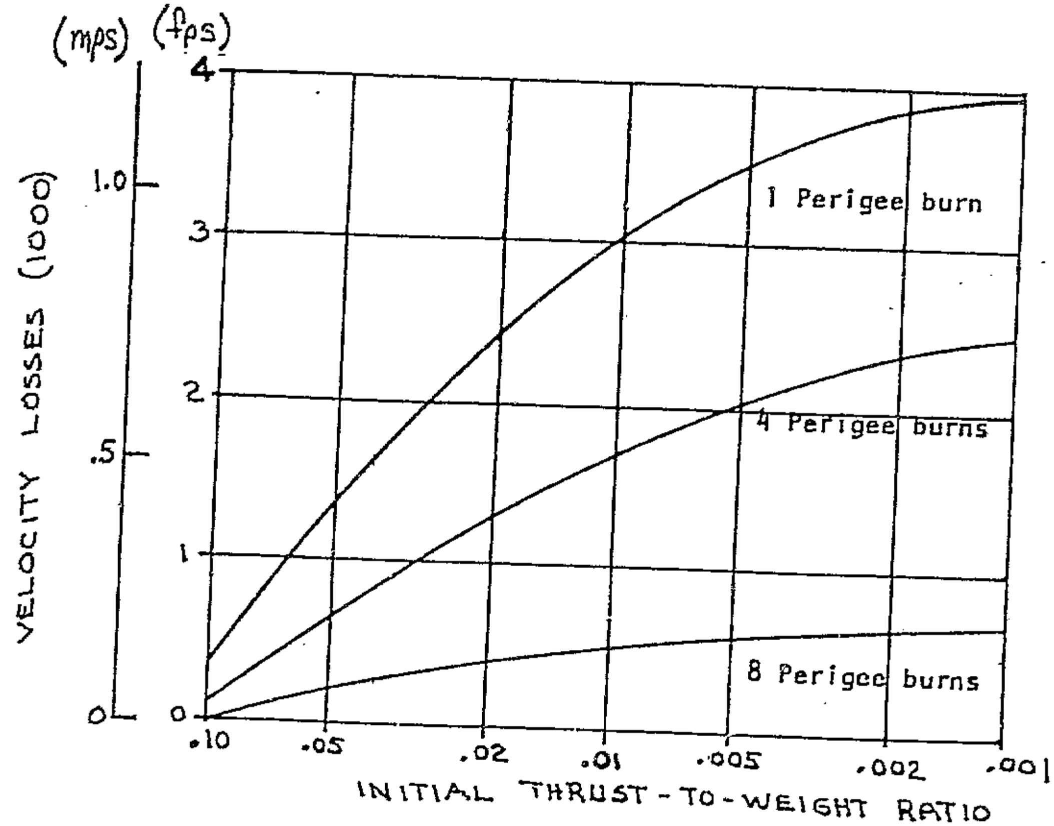


Figure 5.2-34. Velocity Losses for LEO-to-GEO Transfer

The importance of specific impulse and thrust-to-weight ratio on a KSC-launched equatorial geosynchronous mission is illustrated in Figure 5.2-35. The 4270 mps (14,000 fps) mission velocity is representative of impulsive transfer (T/W $\geq$ 1) while the 6000-mps (19,700-fps) figure corresponds to thrust-to-weight ratio less than 10^{-3}. It should be noted that for storable propellants ($313 \leq I_{sp} \leq 340$ sec), the payload-to-initial vehicle gross weight ratio approaches zero (i.e., either there is no payload or the stage weight is infinite) for the higher mission velocities. Thus, extremely low thrust-to-weight missions are not feasible with single-stage storable propellant concepts.

A technique that may be used to decrease the effect of the low thrust-to-weight ratio on the total mission velocity requirement is to perform a multi-perigee burn maneuver. This type of maneuver is illustrated in Figure 5.2-36. Instead of performing a single long perigee burn to raise the apogee to the desired GEO mission altitude, a number of successive perigee burns can be performed, each one raising the apogee by some nominal amount. The effect of this type of maneuver on the required mission velocity is shown in Figure 3.2-34. The velocity loss is the additional velocity that is required above the impulsive (T/W $\geq$ 1) mission velocity as previously expressed in Table 5.2-4. Thus, for example, at an average thrust-to-weight ratio of 0.001 an eight-perigee burn maneuver has the same above impulsive velocity requirement as a single perigee burn mission with an average thrust-to-weight of 0.075. The velocity losses in this instance have been reduced by 1000 mps (3300 fps). Thus, with the multi-perigee burn technique, average thrust-to-weight ratios as low as 0.001 are quite feasible for the storable chemical system since the mission velocity requirement could be reduced to 4270-4570 mps (14,000 to 15,000 fps).

For a large structures platform weighing 60,500 kg (133,400 lb) and a geosynchronous transfer mission velocity of 4270 mps (14,000 fps), a carpet plot illustrating the effects of specific impulse and structural efficiency factor on total propulsion module weight is shown in Figure 5.2-37. Of particular interest is the region representative of storable propellant stages ($0.88 \leq \lambda \leq 0.92$ and $313 \leq I_{sp} \leq 340$ sec). The combination of $\lambda = 0.92$ and $I_{sp} = 340$ sec results in a 218,000-kg (480,000-lb) propulsion module size. At the more conservative values of $\lambda = 0.88$ and $I_{sp} = 313$ sec, the propulsion module size has grown to 349,000 kg (770,000 lb). These two propulsion module concepts, plus a third version representing a midpoint value, will be further evaluated as to the number of Shuttle flights required for the various propulsion module delivery modes.

The effect of the mission velocity on the storable propulsion size is illustrated in Figure 5.2-38. A carpet plot, relating the storable propellant characteristic region for various mission velocities, is shown. Thus, for example, the two higher propulsion module weights chosen for further analysis and discussion can also represent missions with greater than 4270 mps velocity. However, as shown in Table 5.2-5, the structural efficiency and specific impulse values would be different.

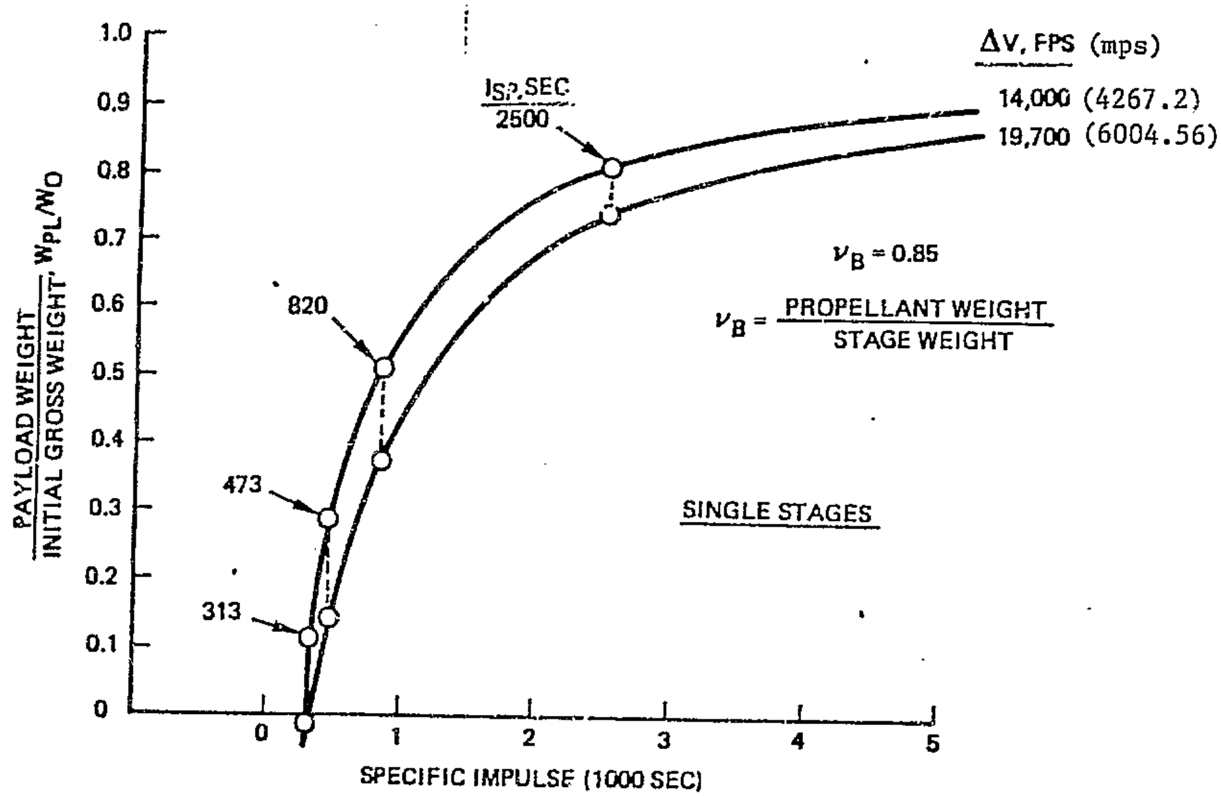
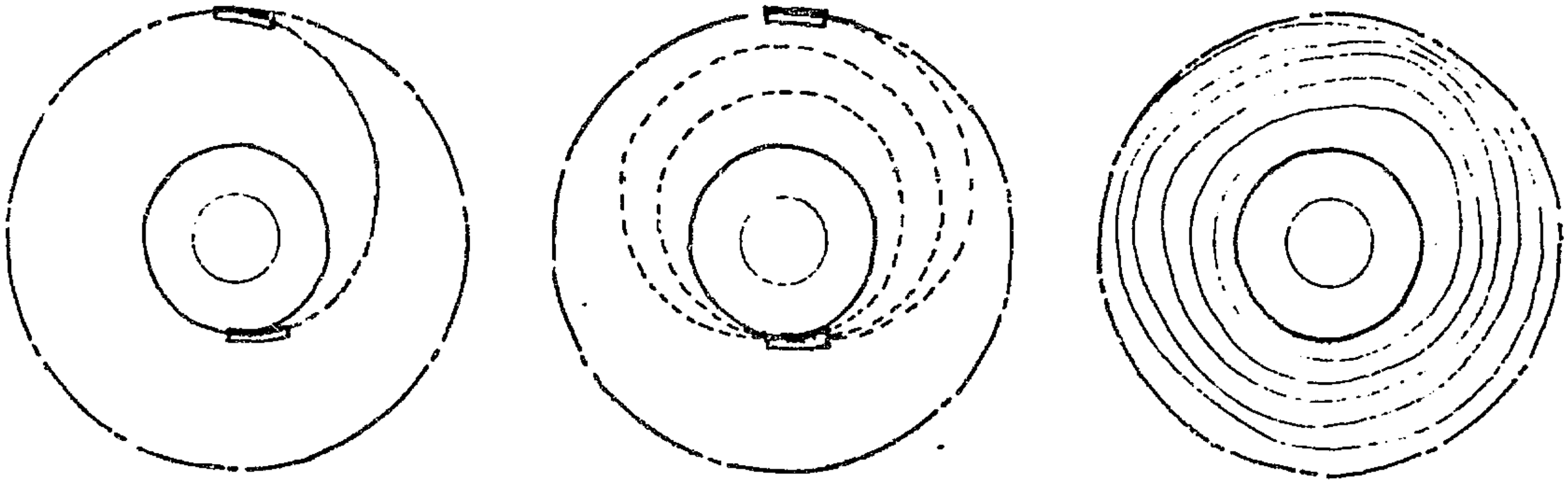


Figure 5.2-35. Effect of Specific Impulse



TWO-IMPULSE

ONE PERIGEE BURN

ONE APOGEE BURN

MULTI-IMPULSE

MORE THAN ONE PERIGEE BURN

ONE INSERTION BURN AT FINAL APOGEE

CONTINUOUS BURN SPIRAL TRAJECTORY

Figure 5.2-36. Types of Transfer Maneuvers

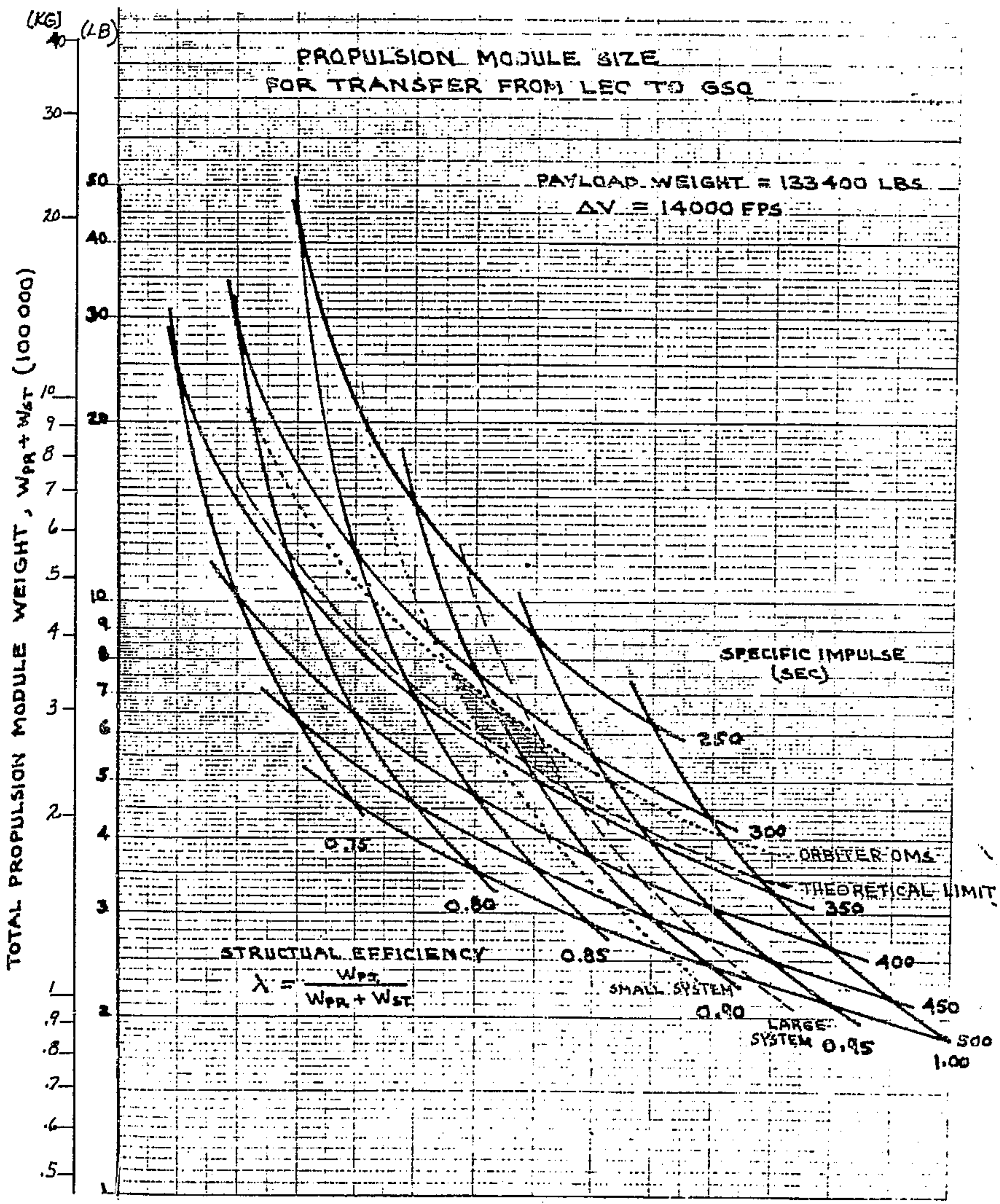


Figure 5.2-37. Parametric Sizing for Advanced Storable Propulsion

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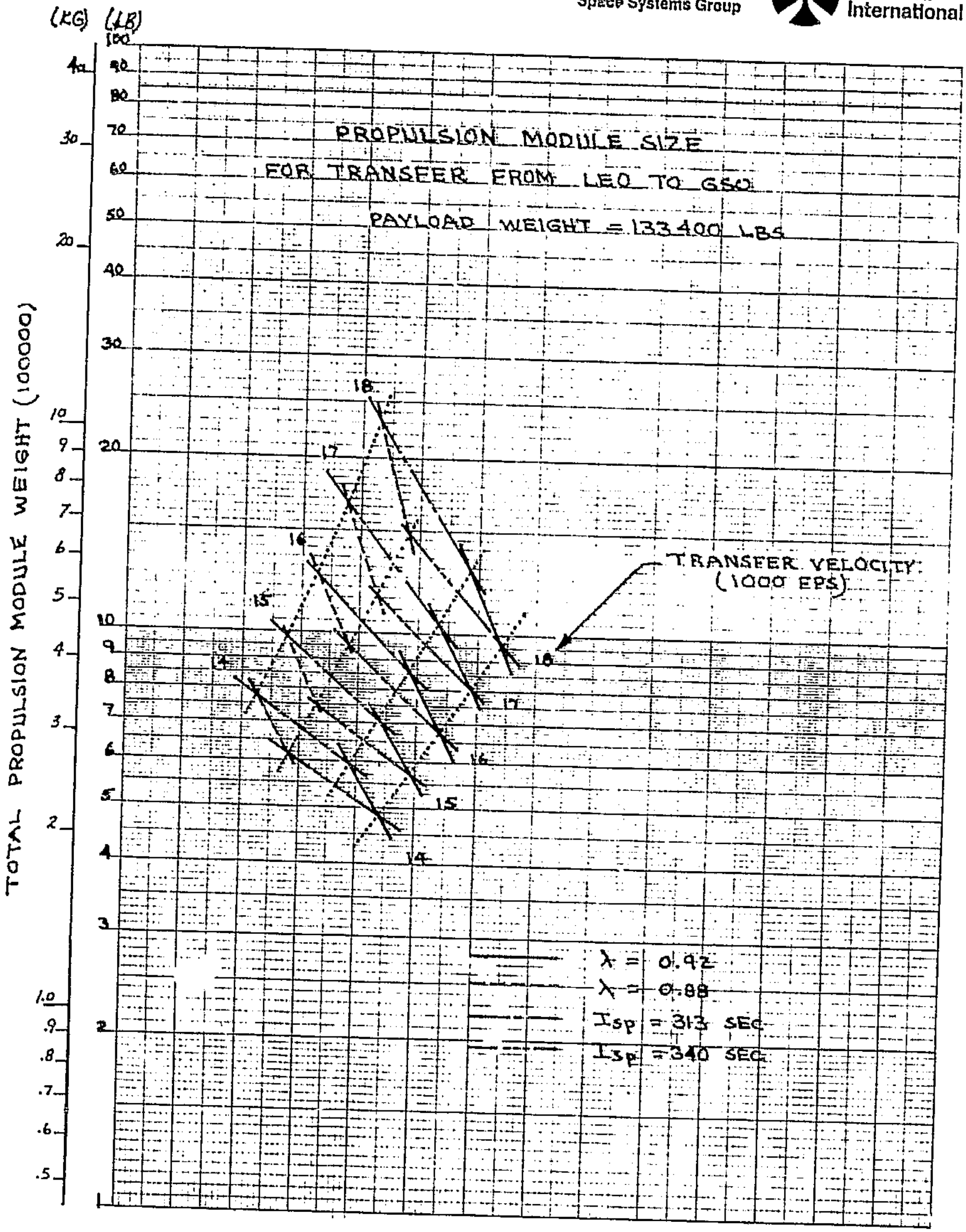


Figure 5.2-38. Delta-V Effects on Parametric Sizing for Storable Propulsion

Table 5.2-5. Advanced Storable Propulsion System Size

Propulsion Module Weight		Primary Mission		Potential Alternate Mission			
		Mission ΔV = 4267.2 mps (14,000 fps)		Mission ΔV = 4572 mps (15,000 fps)		Mission ΔV = 4876.8 mps (16,000 fps)	
kg	(lb)	λ*	I_{sp}	λ	I_{sp}	λ	I_{sp}
217,723.2	(480,000)	0.92	340	–	–	–	–
272,154	(600,000)	0.90	325	0.912	340	–	–
349,264.3	(770,000	0.88	313	0.88	335	0.905	340

* Structural efficiency factor, $\lambda = \dfrac{\text{Propellant weight}}{\text{Propellant weight + structural weight}}$

5.2.2.3 Propulsion Module Delivery Concept Comparison

The various propulsion module delivery concepts that could be used to deliver the necessary propulsion modules to the completed large structure platform in its construction orbit were discussed in considerable detail in Section 5.2-1. However, for reader convenience, the general technique used to calculate the number of Shuttle flights is repeated here.

For the purpose of estimating the number of Shuttle flights required, the propulsion modules, the teleoperator farm, and the super-teleoperator were assumed to be divisible into various but identically sized units, each matched to the Shuttle capability. To estimate the number of units for any particular element (propulsion module, teleoperator, etc.) the total weight of the payload in question was divided by the Shuttle capability to that altitude. The number of flights or units then corresponds to the next highest integer number. Where the payload weight was already evenly divisible by the Shuttle capability, one additional flight was assumed to account for any cradle or other auxiliary equipment requirements.

Thus, for example, a teleoperator farm weighing 29,000 kg (64,000 lb) at a 610-km (330-nmi) altitude would require two Shuttle flights for its delivery, since the Shuttle can deliver and rendezvous at that altitude only a 21,800-kg (48,000-lb) payload. In the comparison of Table 5.2-6, this then would appear as 29,000/2 = 14,500 kg payload weight and 21,800 - 14,500 = 7300 kg as potential auxiliary equipment weight per flight. It is recognized that more optimum payload distributions are possible, particularly where smaller teleoperator deliveries can be combined with other "piggy back" payloads to more completely utilize the Shuttle capability.

Tables 5.2-6, 5.2-7, and 5.2-8 present the Shuttle delivery requirements for storable propulsion concepts ranging from the advanced (I_{sp} = 340 sec) to current technology (I_{sp} = 313 sec). In these three tables, note that the number of flights required to deliver the propulsion modules are listed separately (above for each case) from the number of flights to deliver the teleoperator devices.

For the platform flydown technique, as the number of flights to deliver the propulsion modules increases, additional care should be taken for orbital lifetime calculations. A compromise delivery altitude may be required in which the platform and Shuttle "meet" at a slightly higher, longer-life orbit. This reduces the Shuttle delivery performance, but gives the desired orbit life to assure safe completion of the overall propulsion delivery operation. Another option would be to perform orbit makeup maneuvers with the platform between propulsion delivery flights. Still another possibility would be to deliver some of the propulsion modules prior to completion of the platform construction.

The combined number of Shuttle flights that are required to successfully perform the propulsion module delivery operations are shown in Table 5.2-9 for all of the delivery concepts. Included in this comparison are the data for the cryogenic propulsion modules (from Table 5.2-3) . Since the storable propulsion modules are approximately twice as heavy as the cryogenic

Table 5.2-6. Delivery Requirements for Advanced Storable Propulsion—I_{sp} = 340 sec

ADVANCED STORABLES

I_{sp} = 340 SEC

λ = 0.92

ΔV = 4267.2 mps (14,000 fps)

$W_{PROP\ MOD}$ = 217,723.2 kg (480,000 lb)

	NO. OF SHUTTLE FLIGHTS	PAYLOAD WEIGHT		AUX. EQUIP. WEIGHT		ALTITUDE	
		KG	(LB)	KG	(LB)	KM	(NMI)
DIRECT SHUTTLE DELIVERY	11	19,776.5	(43,600)	1995.8	(4400)	611.16	(330)
SELF-DELIVERY	8	27,215.4	(60,000)	362.9	(800)	611.16	(330)
INDIVIDUAL TELEOPERATORS	8 2	27,215.4 17,962.2	(60,000) (39,600)*	2268.0 5624.5	(5000) (12,400)	351.88 351.88	(190) (190
TELEOPERATOR FARM	8 2	27,215.4 14,514.9	(60,000) (32,000)	2268.0 7257.4	(5000) (16,000)	351.88 611.16	(190) (330
SUPERTELEOPERATOR	8 2	27,215.4 21,318.7	(60,000) (47,000)	2268.0 8164.6	(5000) (18,000)	351.88 351.88	(190) (190)
PLATFORM FLYDOWN	8 1	27,215.4 4,989.5	(60,000) (11,000)**	2268.0 16,782.8	(5000) (37,000)	351.88 616.16	(190) (330)

*FOUR TELEOPERATORS PER FLIGHT

**MAY BE COMBINED WITH A CONSTRUCTION FLIGHT

Table 5.2-7. Delivery Requirements for Advanced Storable Propulsion—I_{sp} = 325 sec

ADVANCED STORABLES

I_{sp} = 325 sec | ΔV = 4267.2 mps (14,000 fps)

λ = 0.90 | $W_{PROP\ MOD}$ = 272,154.0 kg (600,000 lb)

	NO. OF SHUTTLE FLIGHTS	PAYLOAD WEIGHT		AUX. EQUIP. WEIGHT		ALTITUDE	
		KG	(LB)	KG	(LB)	KM	(NMI)
DIRECT SHUTTLE DELIVERY	13	20,955.9	(46,200)	816.5	(1800)	611.16	(330)
SELF-DELIVERY	10	27,215.4	(60,000)	317.5	(700)	611.16	(330)
INDIVIDUAL TELEOPERATORS	10	27,215.4	(60,000)	2,268.0	(5000)	351.88	(190)
	2	17,962.2	(39,600)*	5,624.5	(12,400)	351.88	(190)
	1	8,981.1	(19,800)**	2,812.3	(6200)	351.88	(190)
TELEOPERATOR FARM	10	27,215.4	(60,000)	2,268.0	(5000)	351.88	(190)
	2	18,143.6	(40,000)	3,628.7	(8000)	611.16	(330)
SUPERTELEOPERATOR	10	27,215.4	(60,000)	2,268.0	(5000)	351.88	(190)
	2	26,671.1	(58,800)	2,812.3	(6200)	351.88	(190)
PLATFORM FLYDOWN	10	27,215.4	(60,000)	2,268.0	(5000)	351.88	(190)
	1	4,989.5	(11,000)***	16,782.8	(37,000)	611.16	(330)

*FOUR TELEOPERATORS PER FLIGHT

**TWO TELEOPERATORS PER FLIGHT

***MAY BE COMBINED WITH A CONSTRUCTION FLIGHT

Table 5.2-8. Delivery Requirements for Advanced Storable Propulsion—I_{sp} = 313 sec

ADVANCED STORABLES

I_{sp} = 313 sec

λ = 0.88

ΔV = 4267.2 mps (14,000 fps)

$W_{PROP\ MOD}$ = 349,264.3 kg (770,000 lb)

	NO. OF SHUTTLE FLIGHTS	PAYLOAD WEIGHT		AUX. EQUIP. WEIGHT		ALTITUDE	
		KG	(LB)	KG	(LB)	KM	(NMI)
DIRECT SHUTTLE DELIVERY	17	20,547.6	(45,300)	1,224.7	(2,700)	611.6	(330)
SELF-DELIVERY	13	26,852.5	(59,200)	635.0	(1,400)	611.16	(330)
INDIVIDUAL TELEOPERATORS	13	26,852.5	(59,200)	2,630.8	(5,800)	351.88	(190)
	3	13,471.6	(29,700)*	4,218.4	(9,300)	351.88	(190)
	1	17,962.2	(39,600)**	5,624.5	(12,400)	351.88	(190)
TELEOPERATOR FARM	13	26,852.5	(59,200)	2,630.8	(5,800)	351.88	(190)
	3	14,651.0	(32,300)	7,121.4	(15,700)	611.16	(330)
SUPERTELEOPERATOR	13	26,852.5	(59,200)	2,630.8	(5,800)	351.88	(190)
	3	22,815.6	(50,300)	6,667.8	(14,700)	351.88	(190)
PLATFORM FLYDOWN	13	26,852.5	(59,200)	2,630.8	(5,800)	351.88	(190)
	1	4,989.5	(11,000)***	16,782.8	(37,000)	611.16	(330)

*THREE TELEOPERATORS PER FLIGHT

**FOUR TELEOPERATORS PER FLIGHT

***MAY BE COMBINED WITH A CONSTRUCTION FLIGHT

Table 5.2-9. Delivery Concepts Comparison for Advanced Storable Propulsion Systems

NUMBER OF SHUTTLE FLIGHTS—CONSTRUCTION ALTITUDE, 610 KM (330 NMI)

	CRYO	ADV. STORABLES		
	PROPULSION MODULE WEIGHT	PROPULSION MODULE WEIGHT		
	144,014.8 KG (317,500 LB)	217,723.2 KG (480,000 LB)	272,154 KG (600,000 LB)	349,264.3 KG (770,000 LB)
DIRECT SHUTTLE DELIVERY	7	11	13	17
SELF-DELIVERY	6	8	[illegible]	13
INDIVIDUAL TELEOPERATORS	7	10	[illegible]	17
TELEOPERATOR FARM	6	10	12	16
SUPERTELEOPERATOR	6	10	12	16
PLATFORM FLYDOWN*	6	9	11	14

*ONE OF THE FLIGHTS MAY BE COMBINED WITH A CONSTRUCTION FLIGHT modules, they (the storables) require approximately twice as many Shuttle flights to accomplish the delivery.

The sensitivity to the types of delivery technique increases with both the total weight of the propulsion modules and delivery orbit altitude (construction orbit). For example, for a total propulsion module weight of 14,500 kg (320,000 lb), or less, there exists only a single flight difference between the various delivery techniques at h = 610 km (330 nmi). At the 363,000-kg (800,000-lb) level, the difference is four Shuttle flights.

It can also be noted that the direct Shuttle delivery and the individual teleoperator delivery techniques consistently require more Shuttle flights. The self-delivery and the platform flydown techniques always require the fewest number of Shuttle flights.

5.2.3 Solid IUS Propulsion Concept

Although not usually considered a prime candidate for use with light weight, large area space structures, an IUS based solid rocket propulsion concept was briefly investigated as part of the orbit transfer analysis. According to current plans the IUS will be an operational stage somewhat before the 1990 time period of interest to the study. Thus, it represents an available, fully developed propulsion stage. This availability and attendant low-cost potential prompted this feasibility analysis.

5.2.3.1 IUS Motor Features

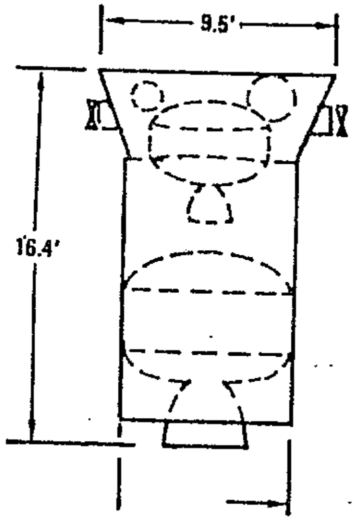
Figure 5.2-39 summarizes the important physical features and performance characteristics of the solid rocket motored IUS. Examination of the performance characteristics for the large and small motors shows the small motor to produce almost 60 percent less peak thrust, thereby making it potentially more attractive for boosting the relatively fragile light weight structure associated with space constructed systems. However, while the thrust level of the small motor is attractive from a T/W standpoint its relatively small propellant load (2720 kg or 6000 lb) compared to the large motor (9710 kg or 21,400 lb) makes it unattractive as an element in the orbit transfer system. For space projects of the size class considered in the study (or larger) it would require a prohibitively large number of small motors as "final" stages to gain any real reduction in peak T/W's. Thus, the IUS derived propulsion systems considered here are based on the use of the large motor.

As shown in the figure these motors may be off-loaded up to 50 percent in order to match mission ΔV requirements. This provides a continuous relationship between ΔV, number of stages and percent off-load. If less than 50 percent off-load were provided gaps would appear in the ΔV versus payload curve where, for example, one stage could not provide enough ΔV and two stages off-loaded to the maximum would provide too much ΔV.

This figure, then, summarizes the important characteristics of the solid motor "building block" used in synthesizing IUS based orbit transfer propulsion systems.

5.2.3.2 IUS Propulsion System Considerations

Figure 5.2-40 depicts an IUS derived orbit transfer propulsion system made up of the large solid motor building blocks described above. Propellant off-loading will be required to match propulsion performance with mission ΔV requirements. Two off-loading possibilities exist, (1) uniform off-loading of all motors used in the perigee burn and different but uniform off-loading of all motors used in the apogee burn, and (2) off-loading only the final burn motors used in the perigee and apogee ΔV's. Use of the "uniform" off-loading concept will reduce the number of motor loading conditions required in the system inventory. If only the final burn motors for each ΔV impulse (perigee and apogee) were off-loaded, then, full stages would be required in addition to the two different off-loaded configurations. Thus, the uniform off-loading concept requires only two loading conditions instead of three, which should reduce propulsion system integration costs.

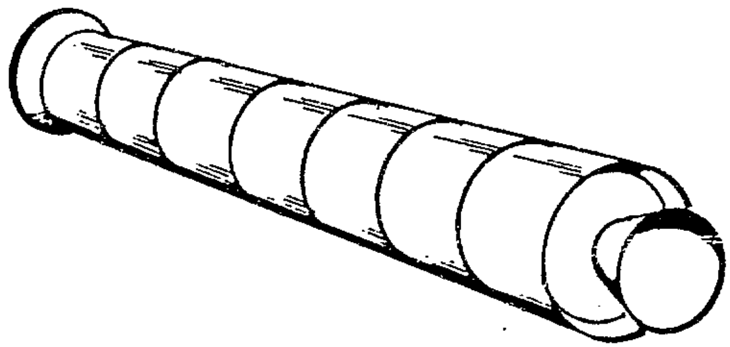


TWO STAGE CONFIGURATION

	LARGE MOTOR	SMALL MOTOR
MAX THRUST, LB	62090	25800
AVG THRUST, LB	43455	17540
I_{SP}, SEC	295.4	294.3
BURN TIME, SEC	145.4	100.4
PROPELLANT WEIGHT, LB	21400	6000
DRY WEIGHT W/SUBSYST, LB	2917	2151
TOTAL WEIGHT, LB	24317	8151
DIAMETER, IN	91	63
LENGTH, IN.	117	75

- CURRENTLY IN DEVELOPMENT, WILL BE AVAILABLE
- OTHER VERSIONS INCLUDE: TWIN STAGE, THREE STAGE & FOUR STAGE
- 50% PROPELLANT OFF-LOAD CAPABILITY
- SINGLE GIMBALLED NOZZLE, EA STAGE
- SECOND STAGE HAS RCS, G&C, AVIONICS & EPS

Figure 5.2-39. IUS Motor Features



- ΔV = 14000 FPS, ONE WAY GEO
- BURN TIME 9 MINUTES PERIGEE, ≈ 3 MINUTES APOGEE
- T/W 0.8 TO 1.4
- PROPELLANT OFF-LOADED FOR PERIGEE & APOGEE ΔV REQ
- TVC/STEERING: SINGLE OMNI-AXIAL GIMBALLED NOZZLE, ±4 DEG
- PROPULSION "STACK" LOCATED ON LONGITUDINAL AXIS THROUGH CG
- AFT & FORWARD SKIRTS PROVIDE INTERSTAGE STRUCTURE

Figure 5.2-40. Example Solid Motor Stacking Arrangement

The number of motors required as a function of weight transported to GEO is shown in Figure 5.2-41. This curve represents the theoretical minimum number of motors required. Depending upon staging, motor stacking arrangements, and off-loading match-ups with perigee/apogee ΔV's the actual number of motors could be as much as 2 or 3 more than the number indicated by the curve.

Based on a platform weight of 60,500 kg (133,000 lb) at least 24 large solid motors would be needed. It is unlikely that such a large number of motors could be integrated into a single long stack. The use of multiple stacks appears to be more promising. Parallel firing of dual stacks would nearly double the T/W ratio, but could provide 3-axis steering while avoiding some of the stage integration problems associated with extreme lengths. With a single stack an additional RCS type system would be required to provide roll control. The use of four stacks, firing only two at a time, would probably be better. It would further simplify the length problem and at the same time preserve the three-axis steering capability. Dual firing would produce values of (T/W) max. approaching 0.9. This is approximately 4 to 5 times greater than the optimum for cryo stages and would result in increased platform weight and in turn more propulsion system weight. System weight penalties (over that for cryo concepts) would fall in the range of 3000 to 5000 kg (6000 to 10,000 lb) as estimated from Figure 5.2-6. More important than the weight impact, T/W's above the 0.3 range will introduce complexities in the local structural arrangements for mounting/installation of large modules. They will also require thicker cap gages than are possible in current beam machine concepts.

The use of odd numbers of stacks (3, 5, etc.) could reduce the peak T/W to values between 0.4 and 0.5. This could be achieved by employing symmetrical stack arrangements around a central stack position. The outer stacks could be fired in parallel pairs and the final burn performed by the central stack to minimize peak T/W's. This would soften the T/W impacts and further reduce the length requirements. However, it would not eliminate the cap gage problem and would require the supplemental RCS system for roll control during the single motor firing intervals.

Thus, stack arrangements which are technically feasible could probably be developed, but they would pose challenging problems to both the structural design of the platform project system and to the beam machine development. Corresponding challenges could be expected for erectables, probably in member sizing. Increases in structural member sizes as well as increases in the number of member types are likely, both of which could affect packaging and construction logistics.

In addition to complications in the design and construction of the project system the large number of modules to be delivered will increase operational costs. The size characteristics of the IUS large motors are not optimumly matched to the shuttle delivery performance. The Shuttle can carry more than two motors, but not three. Thus, delivered in pairs at least 12 Shuttle flights would be required, double that needed for the delivery of cryo stages.

Overall, the single main attribute of the IUS propulsion concept, its existing development and availability must be weighed against the following considerations to determine its applicability to the LEO to GEO transport of large area space systems.

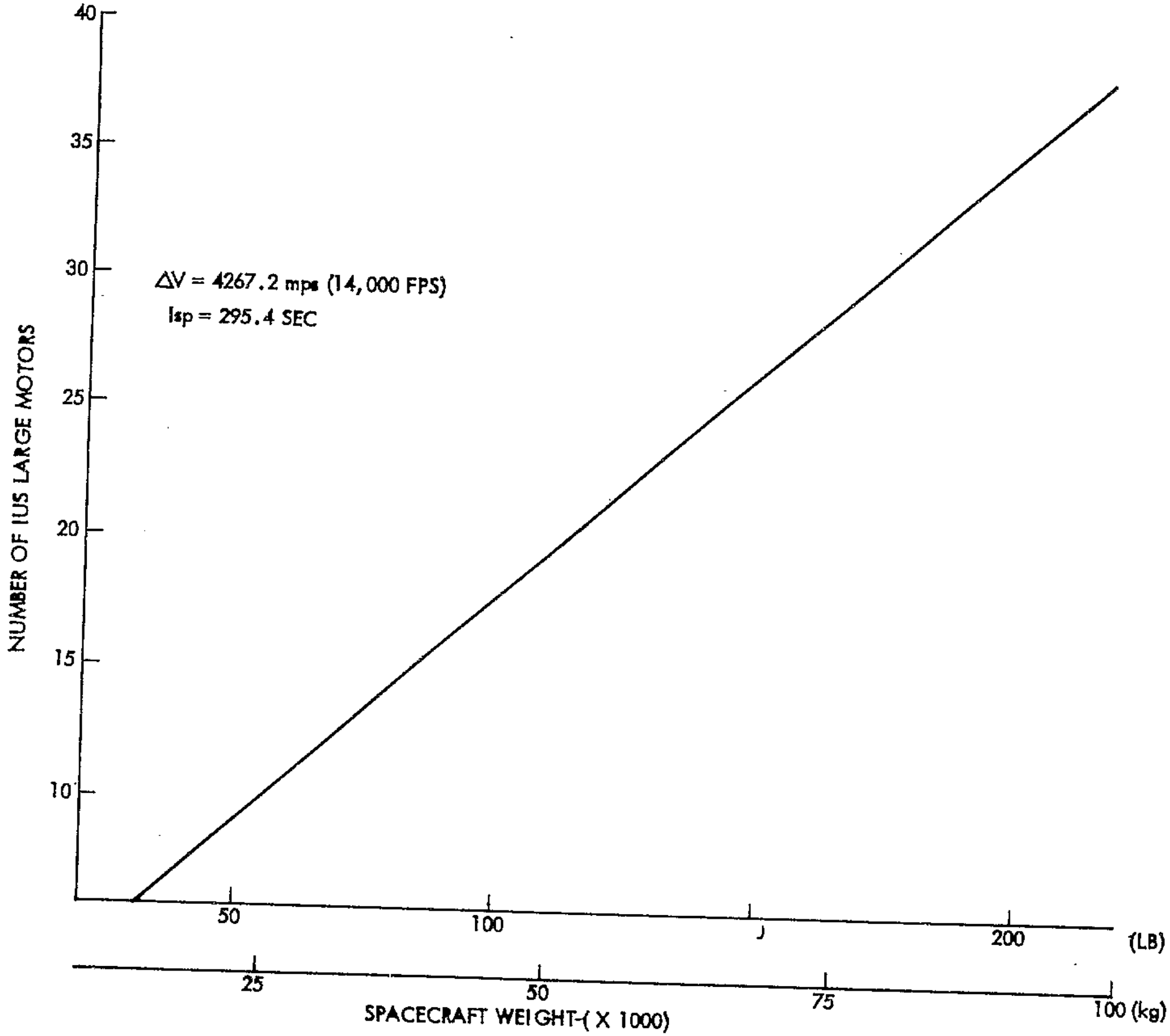


Figure 5.2-41 Number of IUS Large Solid Motors Required Vs. Spacecraft Weight

1. Complexities in the design and construction of the project system.

2. Possible supplemental developments in the construction equipment, i.e., beam machine cap gages.

3. Increased operating costs due to the need for more Shuttle delivery flights.

4. Stack requirements beyond the current two motor units will require development funding which will partially erode the development cost advantage of the IUS over a new cryo or storable stage.

Thus, although the use of IUS solid motors is probably technically feasible on an individual project basis, particularly for projects somewhat smaller than the platform system considered here, its cost effectiveness as a basic LEO to GEO transportation element for large space systems is doubtful. As a general purpose propulsion concept its high operational costs would soon overcome its low procurement cost.

5.2.4 Solar Electric Propulsion

The applicability of very low thrust solar electric propulsion (SEP) to the large structure platform delivery from the low earth construction orbit to the geosynchronous equatorial mission orbit was also briefly analyzed. In particular, the use of 50-cm mercury ion thrusters was evaluated for this mission. Sensitivities as to argon and xenon thrusters were also evaluated.

The chief factors related to missions using solar electric propulsion systems are the trip time and the electric power requirements. The latter could easily exceed subsequent GEO mission requirements. The trip time, on the other hand, could be excessively long from both mission and economic standpoints. However, the comparatively smaller propulsion system weight is highly attractive.

5.2.4.1 Technology Considerations

The employment of solar electric thrusters for transfer of payloads between LEO and GEO in the 1985 to 1995 era is envisioned as a two- or three-step transition. That is, the propellant technology is expected to evolve from mercury, to possibly xenon, to argon. The reasons are that, although mercury is an attractive propellant with respect to thrust and ease of storage, it is expensive, undesirable with respect to atmospheric pollution, and requires heating for earth orbital missions. Xenon and argon are cryogenic materials but appear to present fewer technical problems. Xenon is non-toxic but expensive; argon is practically free, non-toxic, and very abundant.

Xenon behaves much like mercury, producing a little less thrust for the same input electrical power, as shown in Figure 5.2-42. Argon produces still less thrust per watt, but argon thrusters can be operated at much higher current densities (and, hence, temperatures) and therefore at higher power. For large systems, this considerably reduces the number of thrusters, and therefore, reduces complexity. The grid lifetime question (or problem) must be solved, or the erosion rate empirically determined as a function of beam power, before an operational philosophy can be firmly established. For transporting massive payloads, the philosophy of employing a relatively few, large thrusters at high currents, with sufficient redundancy to compensate shortened thruster lifetimes seems to yield a payoff in payload, reliability, and cost (economics of scale).

Currently, the 30-cm-diameter mercury ion bombardment thruster is the only candidate that can provide primary propulsion for GEO missions in the mid-eighties. As of now, a single unit has been operated continuously for over 300 days before failing in 1975. Continuous operation for two years (17,532 hr) is the NASA goal which may be realized by mid-1981. A 50-cm thruster was tested in the last decade, but high-powered solar arrays were not available. Now, with the help of the Space Shuttle, multi-kilowatt and megawatt arrays will be realized in the 1980's, and the 50-cm thruster will probably be preferred. Even larger thrusters are feasible but currently the molybdenum rolling mills are limited to sheets of 60-cm width. A thruster of one-meter active diameter appears to be a reasonable size for large LEO-to-GEO stages with million-kg payloads in the late 1990's. For a 100,000-kg payload in 1990, a 50-cm thruster is believed to be conservative. It is the size selected here for the large communications platform mission.

Showing the thrust (Newtons)/Watt achievable with 50 cm thrusters for three propellants. The R ratio was varied from 0.3 to 0.8 and three grids were assumed. The power per beam ampere was optimistically assumed to be 150 W/A for Hg, 183 W/A for Xe and 200 W/A for A.

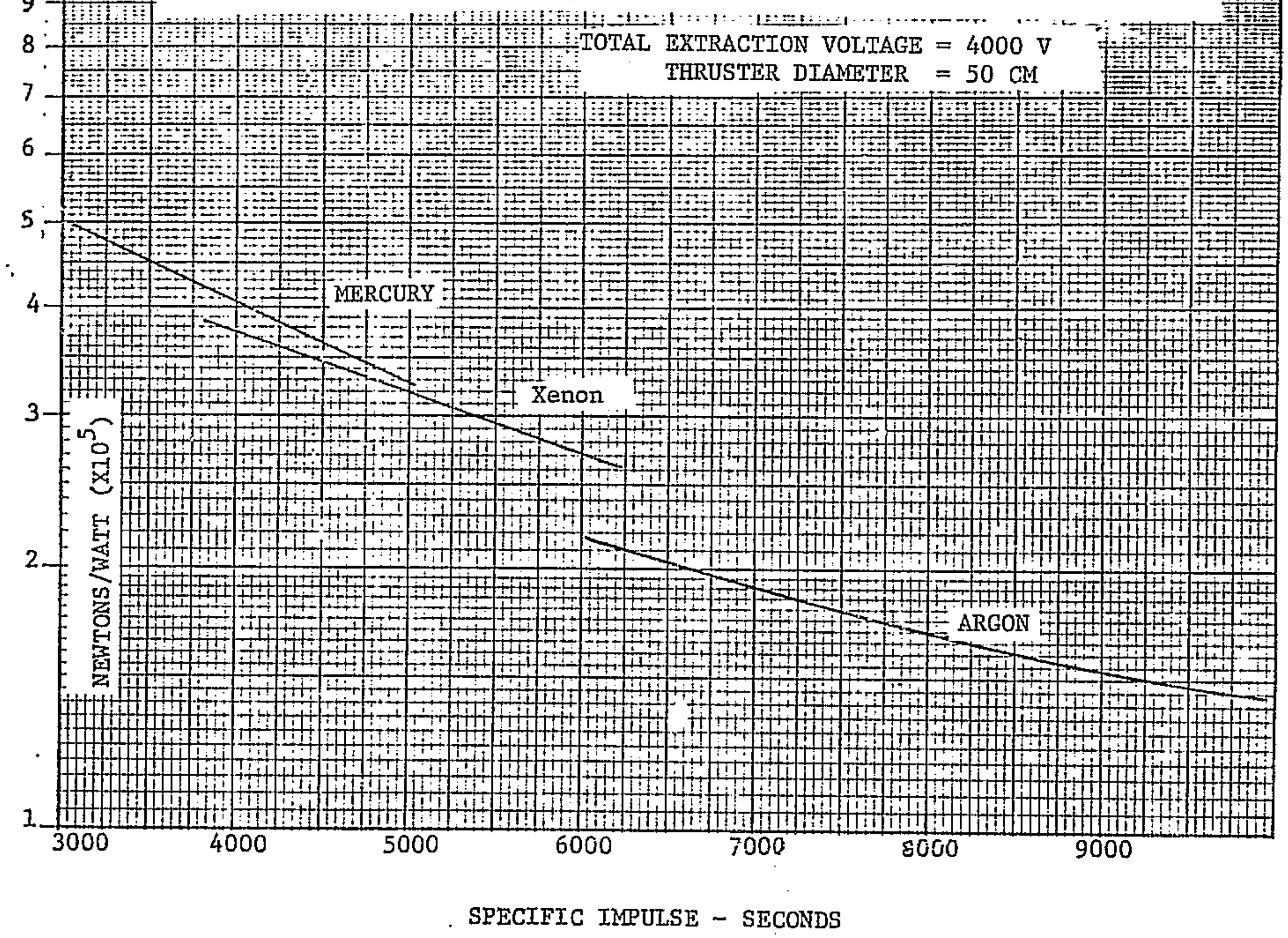


Figure 5.2-42. Achievable Thrust—Mercury, Xenon, Argon Propellants

The benefits derived from technological advances, in general, relate to higher electrical efficiency, coupled with reduced mass, cost, and trip time (Reference 1). When realized, the technological advances can lead to larger thrusters that operate at higher thrust densities (References 1 and 2). This yields a reduced number of thrusters and, therefore, a higher system reliability (Reference 3).

Solar Arrays and Concentrators

For power sources in the 1985-1995 era, gallium aluminum arsenide solar cells, with lightweight solar concentrators, promise significant improvements in primary power sources for electric propulsion concepts. During the early operational period, gallium aluminum arsenide solar cells may become available in quantity and, when employed with solar concentrators, can be made to provide continuous annealing in addition to providing more power. This reduces the average cell radiation damage to no more than approximately 20% and possibly 10% for short trip times. Because of the decrease in cell damage, the solar concentrator, when used with gallium aluminum arsenide cells, could provide the same power as silicon cells with only about 45% of the weight.

The efficiency of silicon (Si) and gallium aluminum arsenide (GaAlAs) solar cells as a function of radiation fluence level is shown in Figure 5.2-43. The results of experiments at Rockwell on GaAlAs solar cells have indicated that for annealing temperatures of approximately 125°C to 150°C, some of the tested cells recover from the degradation effects. Therefore, it may be possible for the cells to operate continuously at temperatures above 125°C to 150°C and experience very little (<15%) radiation degradation. Projected cell efficiencies are shown in Figure 5.2-44 as a function of operating temperature.

In this study, silicon cells without concentrators were used for baseline system sizing.

Power Processor Mass

It appears (Reference 1) that thruster beam and discharge power may be obtained directly from the solar arrays. Since this constitutes about 95% of the power, it represents a considerable savings in system cost and mass through reductions in power processing mass, waste heat radiators, and solar array power (or trip time). In the baseline case considered here, it would eliminate about 77.4% of the power processor mass (saving about 3137 kg).

However, in this study, a more near-term approach was used and the power processor mas was estimated (Reference 2) to be 84.4 kg per thruster for a power processor input power of 35,979 W. This yields 4051 kg for the total array of 48 thrusters.

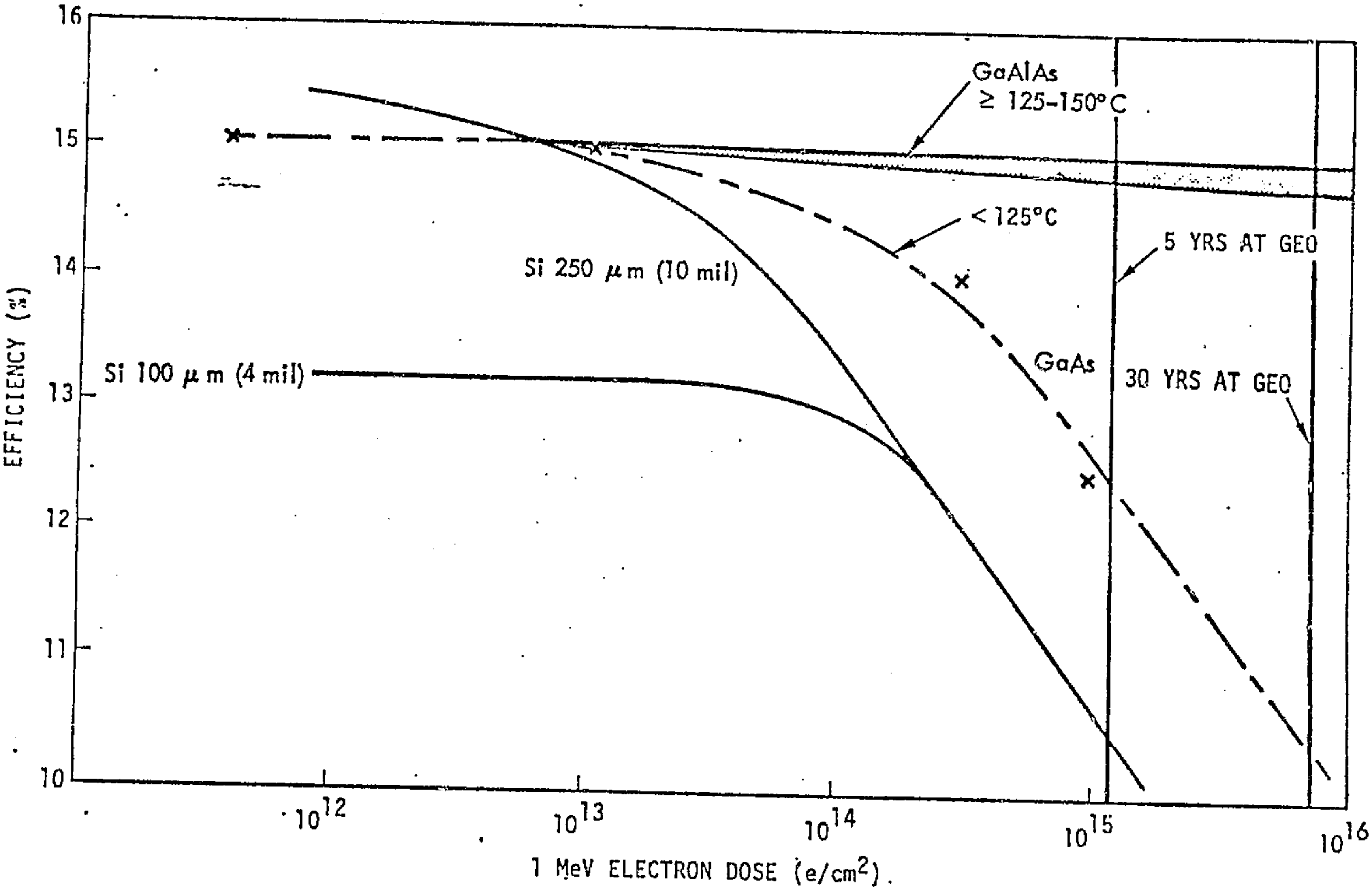


Figure 5.2-43. Experimental Radiation Effects Data

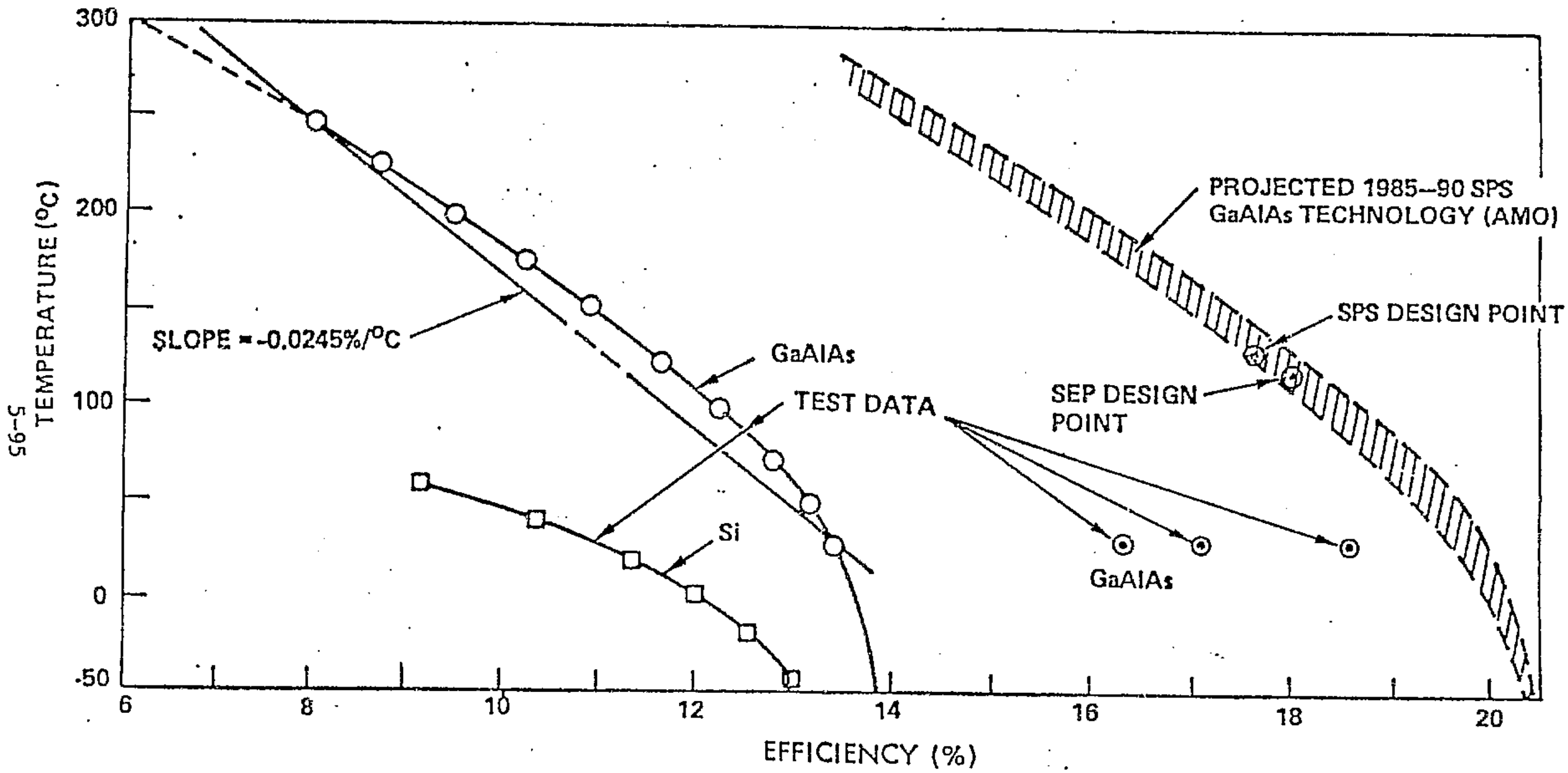


Figure 5.2-44. Solar Cell Temperature Versus Efficiency

Satellite Systems Division
Space Systems Group

Rockwell International

5.2.4.2 Electric Propulsion Subsystem Sizing—Baseline Case

The baseline electric propulsion system discussed here consists of four symmetric arrays of mercury ion bombardment thrusters. The baseline electric power of 1.729 mW is produced by two large solar arrays that operate without the benefit of solar concentrators. Despite this, a payload of 62,000 kg (135,000 lb) can be transported from LEO to GEO in about six months. Sizing of the major subsystem components is outlined below.

Transmission Line Power Losses (see Reference 5)

The power out of the solar arrays going into each thruster array transmission line is 0.43225 mW. Each transmission line is 6 m in length. Using well-insulated aluminum conductor at 323°K, the width, m, and thickness, n, was taken to be

$$m = 4.93 \text{ cm}$$

and $$n = 0.25 \text{ cm}$$

The wire mass M_c for 6m is

$$M_c = 2.56 \text{ kg},$$

and the power loss is

$$P_\ell = 404 \text{ W}$$

Total transmission line mass with heavy insulation is therefore 12 kg. Total power loss is 1616 W.

Electrical losses in the thruster array conductors are no more than 100 W per array. The total conductor power losses from the solar array outputs up to the power conditioners is therefore only 2016 W.

Power Conditioner Losses

P_{ci} = 1,726,984 W total into the power conditioners. Power conditioner efficiency is taken to be 0.88, based on Reference 2. The power out is therefore P_{co} = 1,519,745 W.

Thruster Losses

The electrical losses in the thrusters are largely brought about by discharge losses. Typically they range from 224 to 198 W per beam ampere as the beam current varies from one to four amps (Reference 1). An optimistic value of 150 was used by Byers for a beam current of 13.3 amps (Reference 4). The same figure will be used here since there is a five to ten year lead in the state of the art. Thruster electrical efficiency is then closely approximated by

$$\eta_t \approx V_B/(V_B + 150)$$
$$\approx 0.8888 ,$$

for $V_B = 1200$ volts

$= $ net accelerating voltage

The total thruster beam power is therefore

$$P_B = J_B V_B = 0.8888\ P_{co}$$
$$= 1{,}350{,}885\ W$$

where J_B = the total beam current for N thrusters.

Thruster Characteristics

Each thruster is assumed to have an active diameter of 50 cm and to have three grids. Front diameter ≈60 cm. Thruster mass is given approximately by

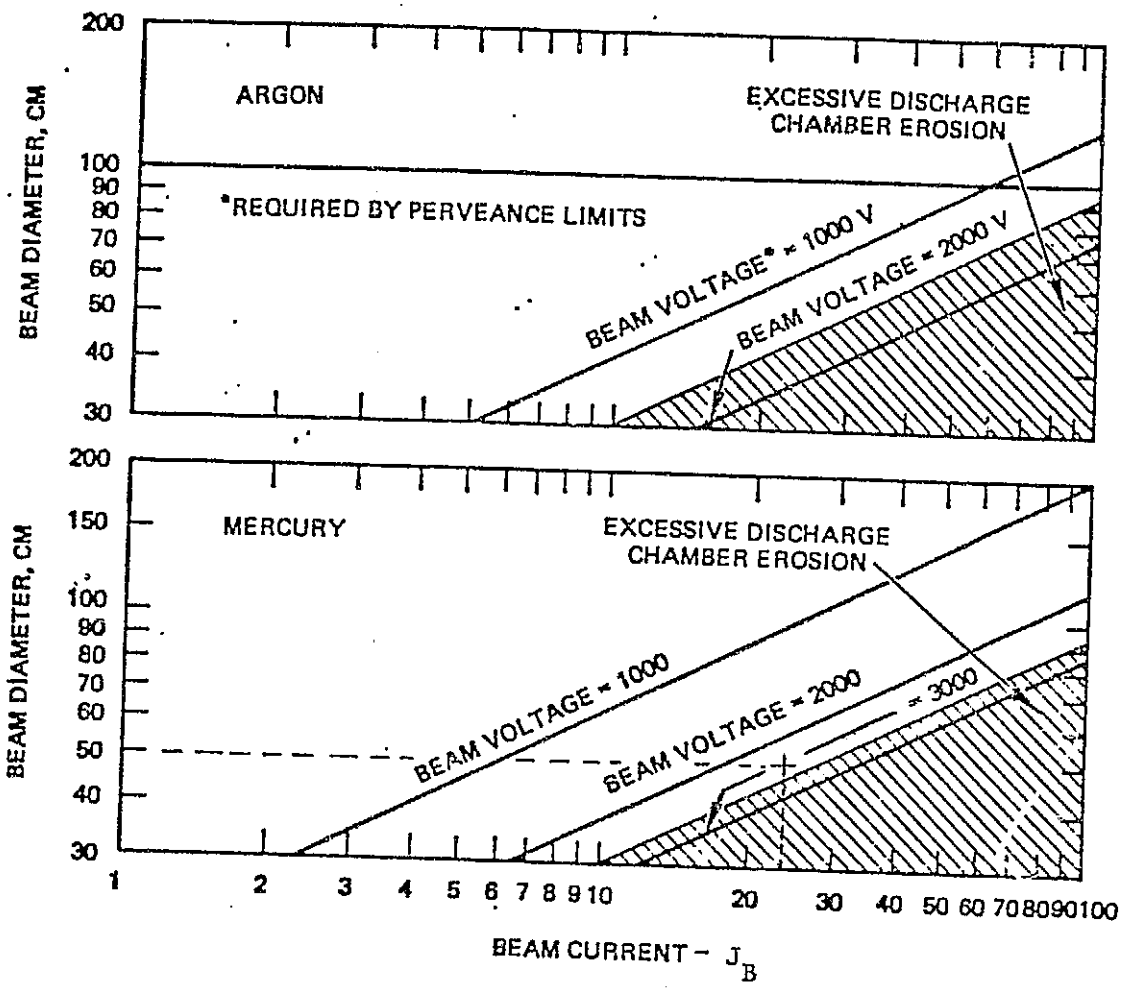
$$M_t \approx 0.1156\ D^{1.35},\ \text{kg}\quad (\text{D in cm}) \qquad (1)$$
$$\approx 22.73\ \text{kg},$$

where the coefficient in Equation (1) was adjusted from an equation in Reference 2 to account for the three grids.

The beam current cannot be so high that excessive grid erosion takes place, thereby excessively shortening thruster lifetime. The plots in Figure 5.2-45 show regions where excessive erosion occurs for mercury and argon thrusters. The beam power per thruster, based on 48 thrusters operating at mission initiation, is 28,143 W. Based on a private communication with Vince K. Rawlins, EPL, NASA/LRC, it is not unreasonable for the 1990 era to assume a total extraction voltage of 4000 V. Further a ratio, R, of 0.3 is entirely feasible with a three-grid system as assumed here.

The corresponding grid temperature is 1030 K for Hg. The net accelerating voltage, V_B, was therefore found from the relation

$$V_B = 4000\ R$$
$$= 1200\ V$$



Beam diameter required to produce argon or mercury ion beams, including wearout limits. The selected operating point is shown by a dashed line. The above diagrams are based on Ref. 2.

Figure 5.2-45. Beam Diameter Required to produce Mercury or Argon Ion Beams

Beam current is therefore

$$J_B' = 28{,}143/1200$$
$$= 23.45 \text{ amperes,}$$

The total beam current from the four thruster arrays is therefore

$$J_B = 1125.7 \text{ amperes}$$

which is shared by 48 thrusters. The specific impulse is determined by

$$I_{sp} = 1415.55\,\eta_u\sqrt{V_B/M} \tag{2}$$

where M = molecular weight

and $\eta_u = 0.94$ propellant utilization

A beam divergence loss of 0.95 is used in most of the references which further reduces I_{sp}. Equation (2) then becomes

$$I_{sp}' = 1344.77\,\eta_u\gamma\sqrt{V_B/M}$$
$$= 3091.8 \text{ sec. (Hg)}$$

In this treatment, I_{sp} will refer to the uncorrected beam (I_{sp} = 3254 sec).

The silicon solar cells degrade with trip time between LEO and GEO. For the case considered here (180 days exposure) the average power is 61% of beginning of life (BOL) power (Figure 5.2-46).

• Thrust

An average thrust, $\overline{F}$, can be used for the mission. Thrust, F, is given by:

$$F = 2P_B\eta_u\gamma/g\,I_{sp}$$
$$= 2 \times 1200 \times 1125.74 \times 0.94 \times 0.95/9.806 \times 3254$$
$$= 75.6 \text{ Newtons}$$

and

$$\overline{F} = 0.61 \times 75.6$$
$$= 46.1 \text{ Newtons}$$

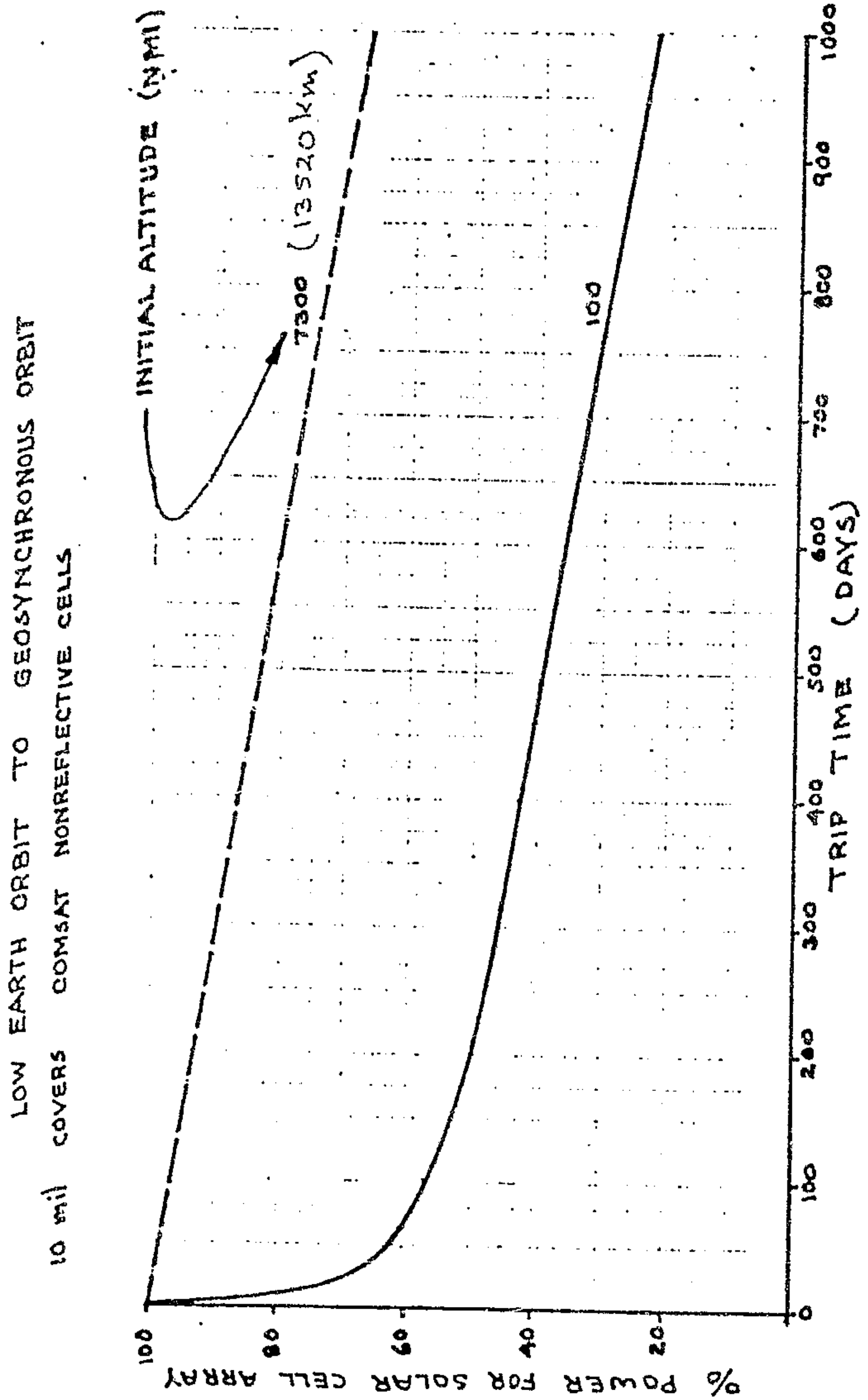


Figure 5.2-46. Solar Cell Array Power Degradation

Also, the average propellant flow rate is given by

$$\dot{M}_p = \frac{\bar{F}}{g\ I_{sp}\gamma} = \frac{46.1}{30313.29}$$

$$= 1.5208 \times 10^{-3}\ \text{kg/sec}$$

• Trip Time

The burnout mass, m_f, was estimated in the parametric study to be 97,740 kg. The propellant required is given by

$$M_p = M_f\ (e^{\Delta V/gI_{sp}\gamma} - 1)$$

where $\Delta\bar{V} = 5{,}969$ m/s, (2 percent margin)

$I_{sp}\gamma = 3091.3$ s,

and $g = 9.806\ \text{m/s}^2$

Thus $Mp = 0.21763\ M_f$

$= 21{,}271$ kg

Thrust duration $= 1.1571 \times 10^{-5}\ Mp/\dot{M}p$

$= 161.9$ days

Trip time $=$ 179.9 days

Total thrust as a function of solar array BOL power is shown in Figure 5.2-47. The average thrust over the mission duration (corresponding to the BOL power) is also shown.

Figure 5.2-48 shows a curve of total trip time (10% in the earth's shadow) versus solar array BOL power. The dashed line shows the selected baseline case.

5.2.4.3 Propulsion System Sizing

The solar electric propulsion systems are characterized by very low average accelerations, i.e., thrust-to-weight ratio less than 10^{-4}. These systems represent the long transfer time continuous burn spiral trajectories illustrated in Figure 5.2-49. From previous analysis, the mission velocity for low earth orbit to geosynchronous orbit transfer with a 28.5-degree plane change has been calculated at 5.97 km/sec (19,580 fps). From a 610-km (330-nmi) altitude, up to approximately 5000 km (2700 nmi) altitude essentially a tangential thrust spiral characterizes the trajectory. Inclination through this phase remains essentially constant (28.5 - 24 degrees). Thereafter, the thrust vector is directed to accomplish more of the plane-change rotation. This flight phase

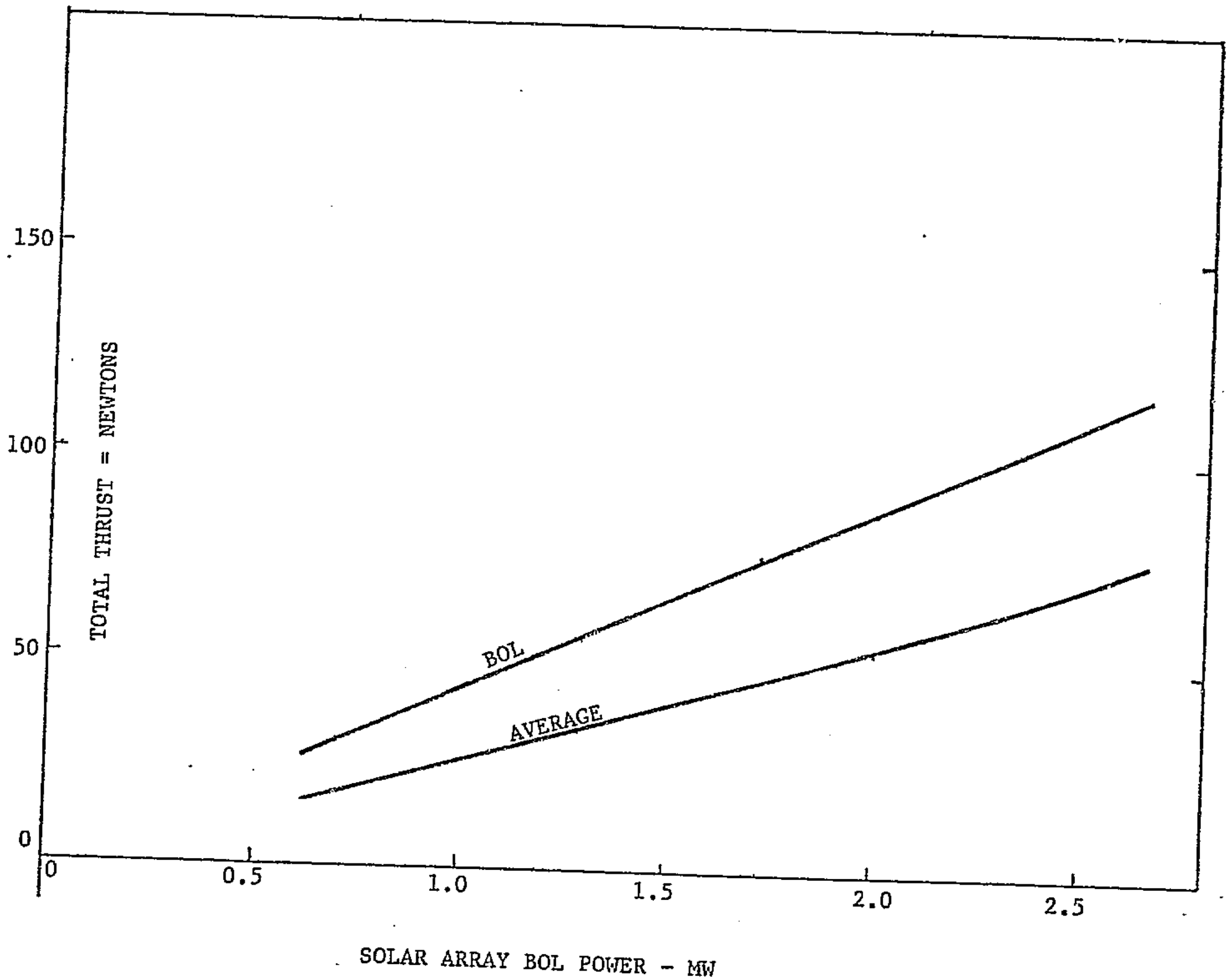


Figure 5.2-47. Total Thrust as Function of Solar Array BOL Power

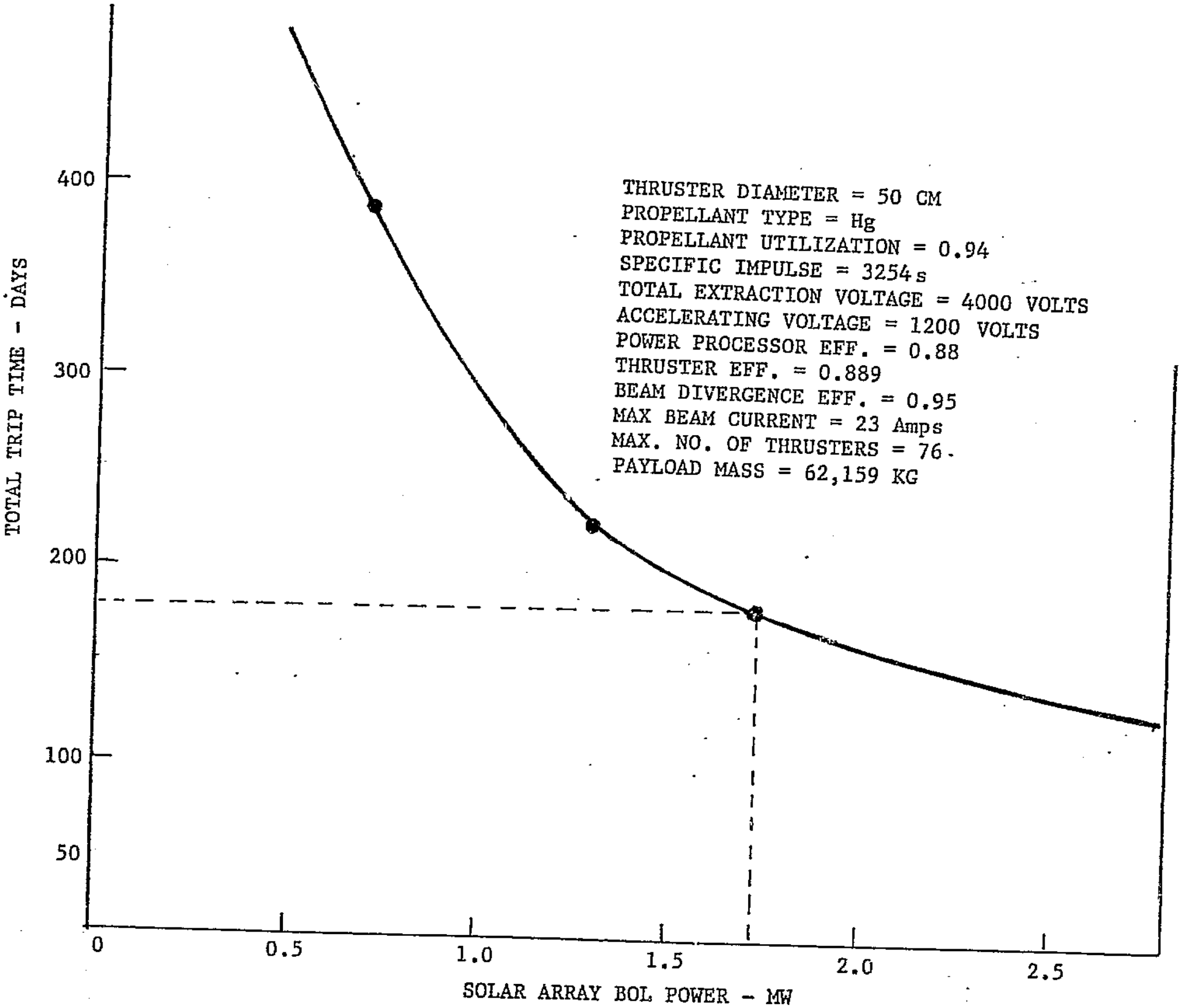
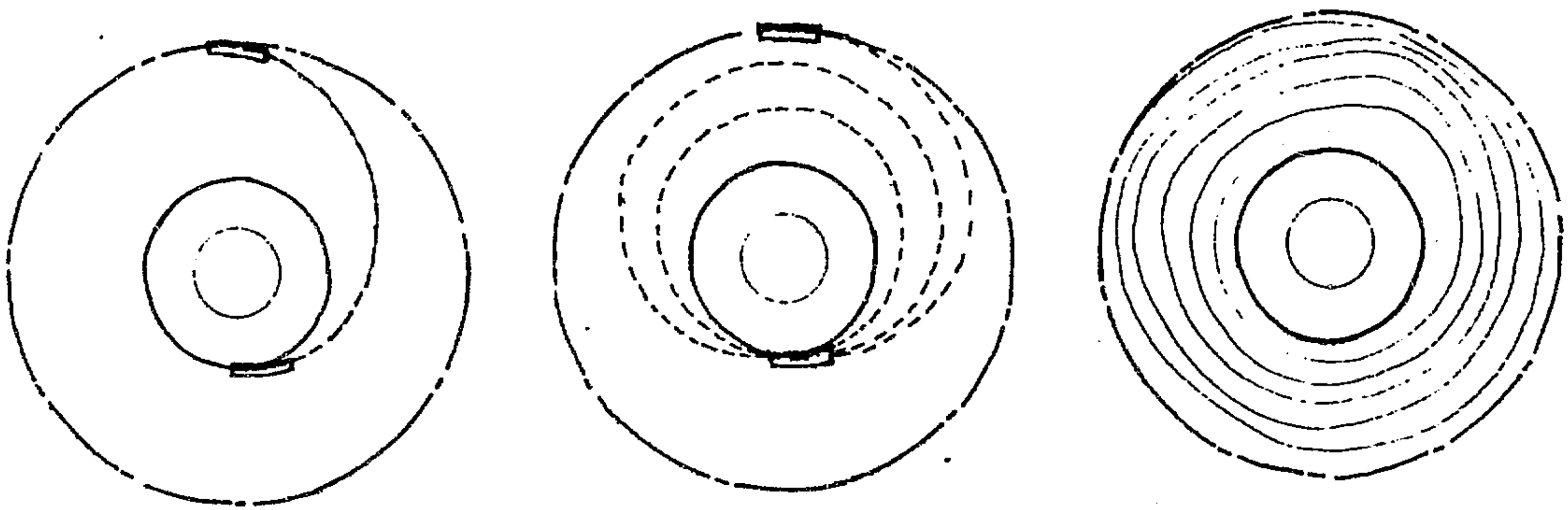


Figure 5.2-48. Total Trip Time Curve

T/W ≥ 0.1
$13{,}500 \leq \Delta V \leq 14{,}000$ FPS
$4114.8 \leq \Delta V \leq 4267.2$ MPS)

0.1 < T/W < 0.001
$13{,}500 \leq \Delta V \leq 18{,}000$ FPS
($4114.8 \leq \Delta V \leq 5486.4$ MPS)

T/W < 0.001
$\Delta V \sim 19{,}600$ FPS
(5974.08 MPS)



TWO-IMPULSE

ONE PERIGEE BURN

ONE APOGEE BURN

MULTI-IMPULSE

MORE THAN ONE PERIGEE BURN

ONE INSERTION BURN AT FINAL APOGEE

CONTINUOUS BURN SPIRAL TRAJECTORY

Figure 5.2-49. Types of Transfer Maneuvers combines the spiral trajectory with the required plane change. A representative spiral trajectory altitude profile is illustrated in Figure 5.2-50. The electrical power required to run the solar electric thrusters is generated by appropriately large solar array system. The power available at any time depends on solar illumination and the efficiency of the solar cells in the array, as discussed in Section 5.2.4.1.

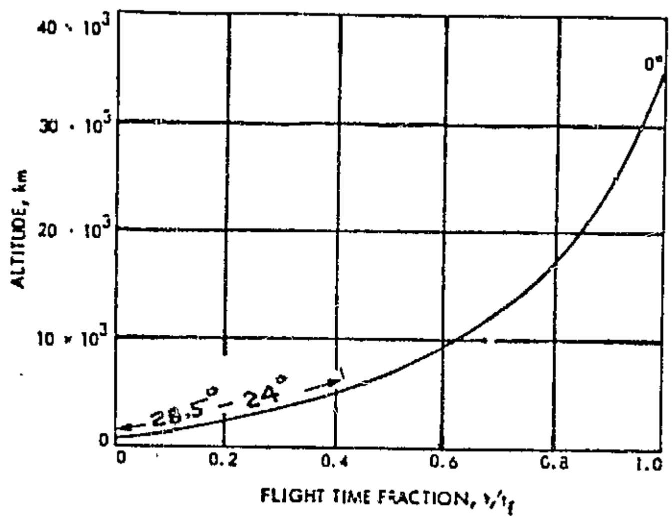


Figure 5.2-50. Spiral Trajectory Profile to GEO

The characteristic of a low inclination spiral is that a considerable portion of each spiral revolution lies in the earth's shadow. The occultation period depends both on the altitude and the relative location of the sun with respect to the orbit plane. The maximum occultation time as a function of altitude is shown in Figure 5.2-51. The maximum occultation time results when the sun is in the orbit plane (sun to orbit plane angle, $\beta = 0°$). At all other β-angles the occultation time is lower. The sun β-angle for orbit plane inclinations of 28.5° or lower cannot exceed 52°. Maximum β equals the orbit inclination plus 23.4°. During occultation periods the SEP cannot be used and, hence, the total trip time by necessity increases. For the purposes of this investigation it is assumed that the trip time increase of 10% over that time, if the entire spiral were to be in sunlight, is typical. This conservative estimate should also account for any thruster preheating requirements before the various power-on flight phases.

The efficiency of the solar cells in the array depends on the length of time the solar arrays spend in the electron and proton radiation belts that must be traversed by the spiral trajectory. For nonreflective silicon cells with 10-mil cover slides the percentage of power-gathering capacity remaining after the transfer to geosynchronous orbit is shown as a function of trip time (Figure 5.2-46). Based on technology considerations, the 50-cm mercury ion thrusters were selected as the baseline solar electric propulsion system for the 1990 time period. The general characteristics of the 50-cm mercury ion thrusters used in subsequent analysis are shown in Figure 5.2-52. System differences introduced by, say, the 100-cm thrusters would be negligible. Thruster grids with very close spacing have only been developed for the 30-cm thrusters. Hence, extrapolating grid sizes larger than 50-cm diameter is still questionable.

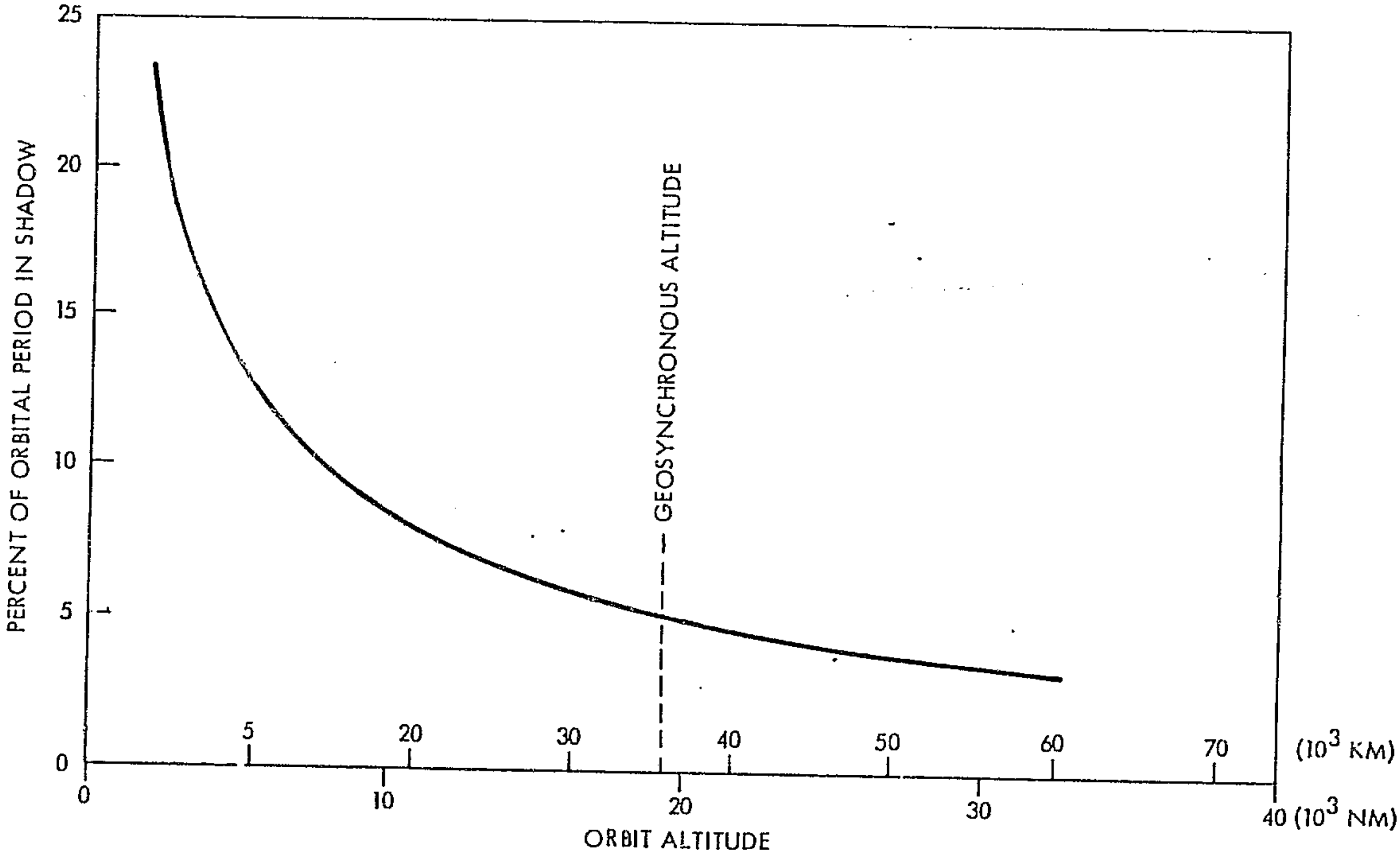


Figure 5.2-51. Maximum Time in Earth Shadow (Circular Orbits)

PROPELLANT TYPE		MERCURY
THRUSTER DIAMETER (ACTIVE)		50 cm
THRUSTER DIAMETER (FRONT)		60 cm
POWER CONDITIONER EFFICIENCY		0.88
THRUSTER ELECTRICAL EFFICIENCY, $\eta_T = \dfrac{V_B}{V_B + 150}$		0.8888
GRID TEMPERATURE		1030°K
ACCELERATOR VOLTAGE	V_A	2800 V
SCREEN (ANODE) VOLTAGE	V_B	1200 V
R (THREE-GRID SYSTEM)	$R = \dfrac{V_b}{(V_G + V_B)}$	0.3
BEAM CURRENT	J_B	23.45 AMPERES
BEAM POWER	P_B	28.140 WATTS
BEAM DIVERGENCE FACTOR		0.95
PROPELLANT UTILIZATION	η_u	0.94
EXHAUST VELOCITY		33.945 km/sec
SPECIFIC IMPULSE (UNCORRECTED BEAM)	I_{sp}	3254 sec
SPECIFIC IMPULSE (CORRECTED)	I_{sp}'	3091.8 sec
THRUST		1.575 N
MASS		22.73 kg

Figure 5.2-52. Electric Thruster Characteristics

The number of mercury-ion thrusters and, hence, the thrust levels that can be developed depend on the power available to the SEP system. This relationship is illustrated in Figure 5.2-53. The BOL specific thrust corresponds approximately 0.044 N/kW.

The average thrust and the end of life (EOL) thrust levels depend on the mission flight time through the radiation belts. The relationships shown correspond to the range of T/W's and flight times required for delivery of the 62,000-kg communication platform to its geosynchronous orbit.

For the total power levels considered (less than 3000 kW), the average thrust-to-weight ratios fall considerably below the 0.001 value. These SEP missions, as discussed previously, then represent the spiral geosynchronous transfer trajectories. The mission delta-V is approximately 5.97 km/sec (19,580 fps). Taking solar array degradation and earth shadowin time into account the trip time from low earth orbit to the geosynchronous orbit is shown in Figure 5.2-54 as a function of the BOL solar array power. The corresponding end of transfer, i.e., the power available at the beginning of the GEO mission is also illustrated.

Reduced trip time means higher thrust electric propulsion systems and, hence, much higher power requirements. The communication platform would also arrive in the geosynchronous orbit with very large excess electrical power capabilities. For example, for a total trip time of six months, the beginning of life power required is 1730 kW and corresponds to approximately 850 kW beginning-of-mission (geosynchronous orbit) power. This value is considerably in excess of the 165 kW BOL power required for the platform. Some excess power, however, could be justified for growth, but such excessive power also represents a much higher cost element to the project.

On the other hand, with the beginning of geosynchronous mission power of only 165 kW, the corresponding trip time to the geosynchronous altitude would be approximately 550 days (18 months). Since the communication platform represents a very large investment, such long trip times could represent a significant delay in revenue generating operations. Hence, the relative cost of excessive power versus potential loss of revenue due to long trip times must be evaluated.

For the SEPS, the mass of the solar arrays, wiring, slip rings, and other electrical items associated with the system above the basic communication payload requirements is shown in Figure 5.2-55 as the SEPS inert weight. The mercury propellant required for the mission is also shown along with the total SEP weight (mercury plus inert) that must be delivered to the communication platform in its LEO construction orbit.

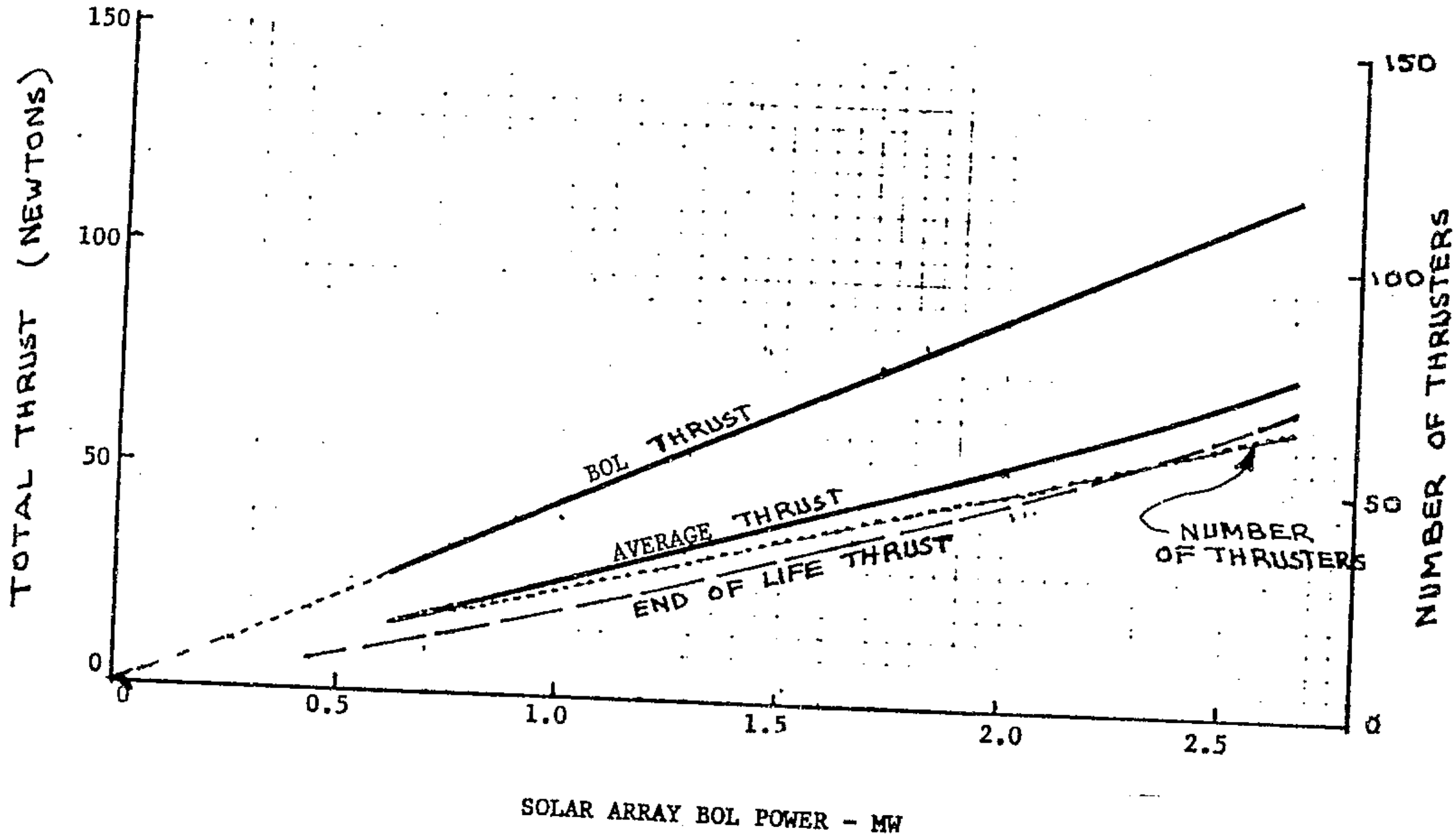


Figure 5.2-53. Thrust Level and Number of Thrusters Relationship

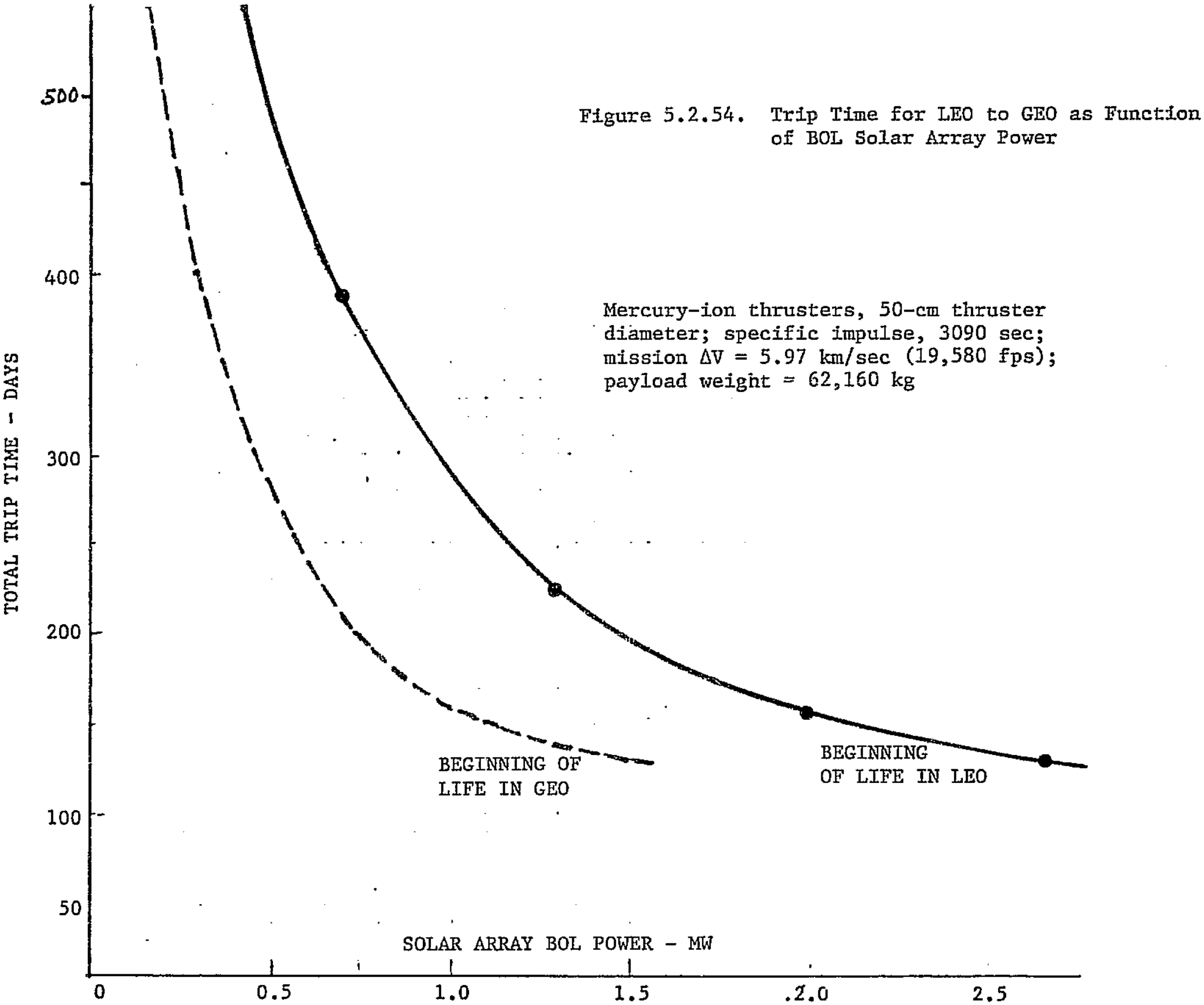


Figure 5.2.54. Trip Time for LEO to GEO as Function of BOL Solar Array Power

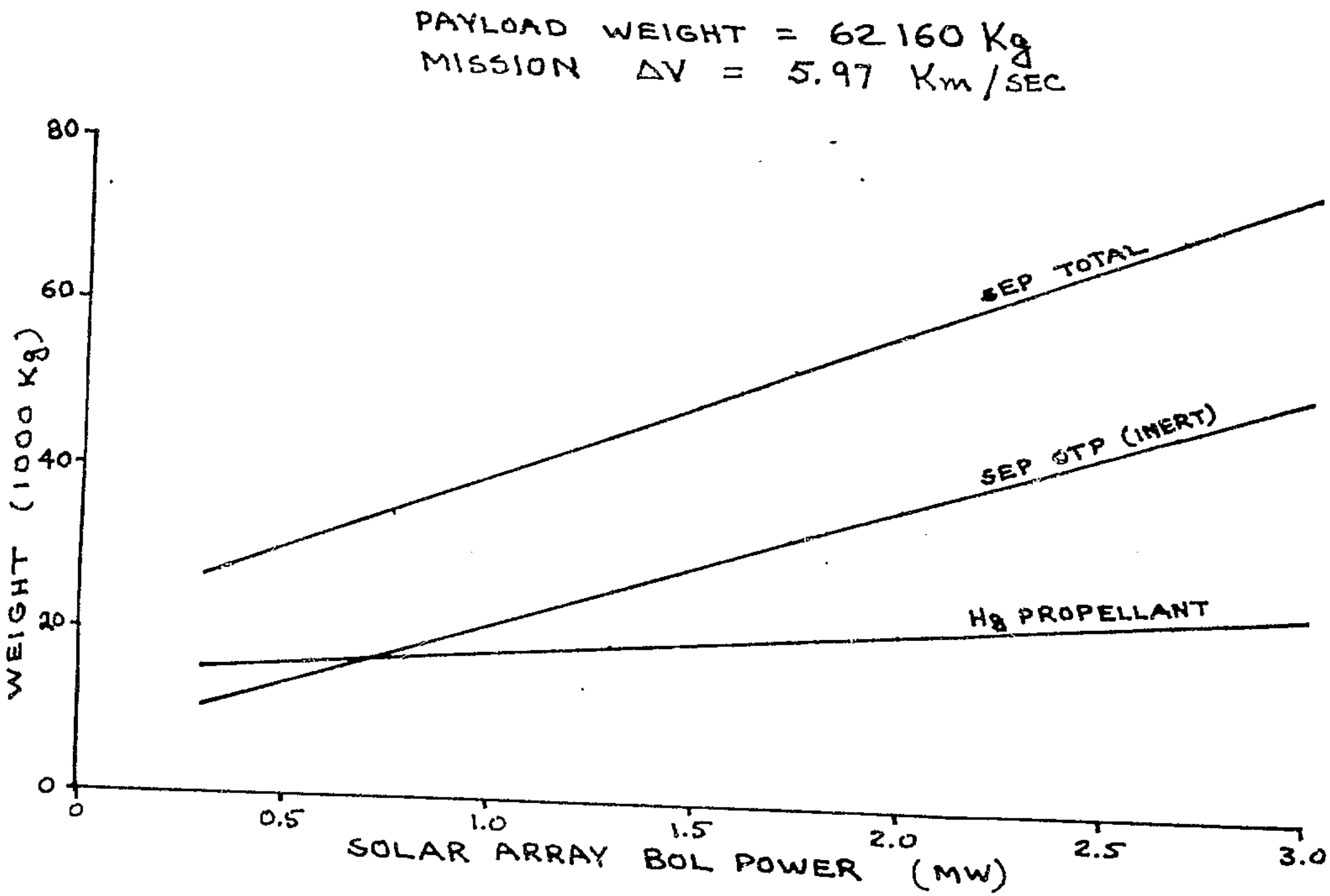


Figure 5.2-55. SEP Weight for LEO-to-GEO Transfer

Since the SEP system postulated here is an integral part of the platform, only the direct Shuttle delivery mode can be considered. An alternative to this concept could be an independent reusable Solar Electric Propulsion Orbital Transfer Vehicle (SEP OTV). The SEPS characteristics and delivered weight to the platform are shown in Figure 5.2-56 for three representative geosynchronous trip times (6, 12, and 18 months). The number of Shuttle flights required for the delivery of the SEP related equipment range from one to three, depending on the construction orbit altitude of the communication platform and the SEPS weight/trip time.

TRIP TIME (MONTHS)	18	12	6
AVERAGE ACCELERATION (10^{-5} G)	1.41	2.12	430
POWER BOL, LEO (KW)	435	760	1660
POWER BOL, GEO (KW)	165	335	845
NUMBER OF THRUSTERS	10	18	40
TOTAL SEP WEIGHT (KG)	29,000	35,000	52,000
DIRECT SHUTTLE DELIVERY			
NO. FLTS TO 611.16 KM (330 NMI) ALTITUDE	2	2	3
NO. FLTS TO 351.88 KM (190 NMI) ALTITUDE	1	2	2

Figure 5.2-56. SEPS Characteristics and Delivered Weight to Platform

5.2.4.4 Hybrid Propulsion

A very cursory analysis was performed for the use of a hybrid propulsion system to deliver the large structure platfrom from a low-altitude construction orbit to geosynchronous orbit. The hybrid system consists of higher thrust-to-weight chemical stages that are used to deliver the platform together with a smaller solar electric propulsion stage to some intermediate changeover altitude. The SEP system then performs the remainder of the mission. This type of flight profile is illustrated in Figure 5.2-57. The main advantage of such a propulsion system combination is the smaller power requirements for the SEP as well as lower chemical propulsion module weights.

The characteristics of the optimum changeover orbit and the corresponding SEP ΔV requirements are shown in Figure 5.2-58 as a function of the velocity that is available from the chemical stages.

For illustrative purposes, a cursory analysis was performed to show the effect of the combined hybrid stage weights on total transfer trip time. The basic single technology propulsion systems that were taken as the corner points are:

- SEP only—365-day trip time—weight = 35,000 kg (77,000 lb)
- Cryogenic stages only—weight = 144,000 kg (317,500 lb
- Storable stages only—weight = 272,000 kg (600,000 lb)

The sizing and delivery of these stages were derived in the previous selections.

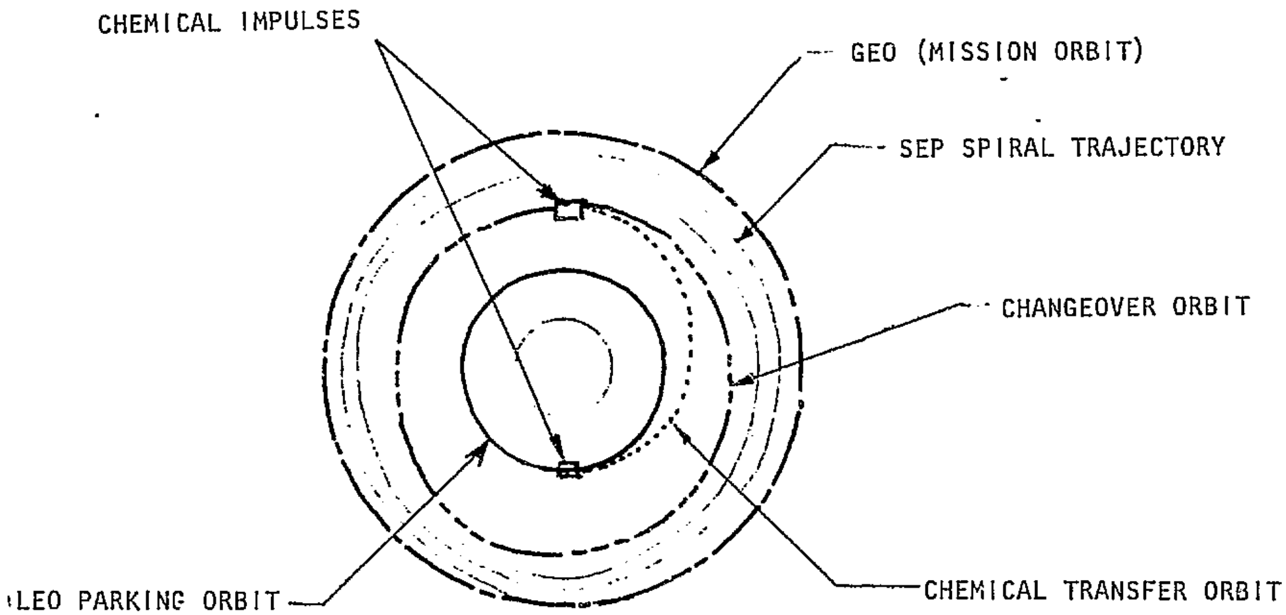


Figure 5.2-57. Hybrid Transfer

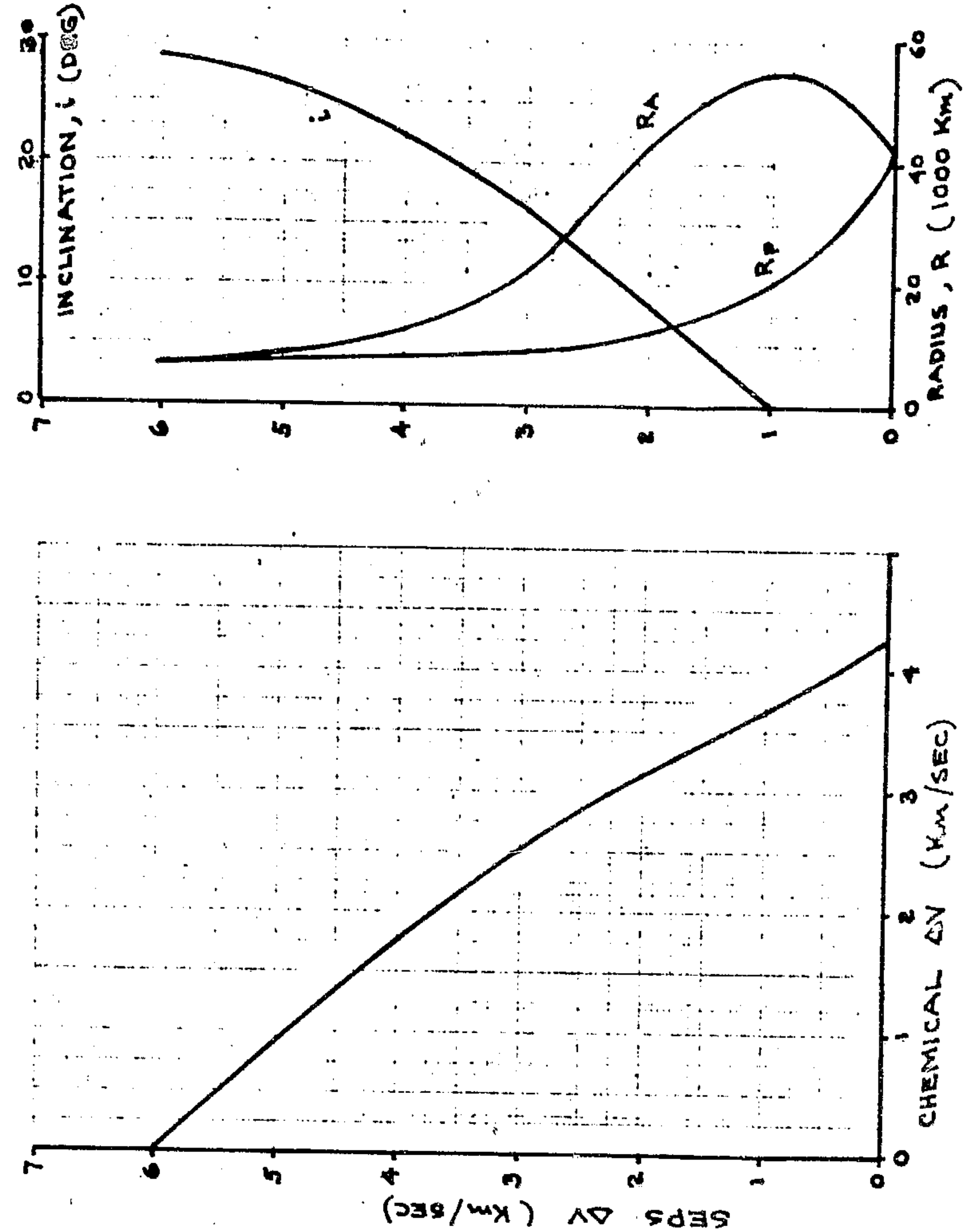


Figure 5.2-58. Characteristics of Changeover Orbit

With the hybrid system it can be assumed that the SEP portion of the trajectory will dominate the trip time calculations because of the extremely low average acceleration of the SEP (2.12×10^{-5} g for the case considered here).

The total hybrid stage weight, including separate chemical and SEP elements, along with the resulting trip time are shown as a function of the mission velocity provided by the chemical stages in Figures 5.2-59 and 5.2-60 for cryogenic and storable chemical stages, respectively. The combined weight shown in these figures is indicative of the total propulsion module weight that must be delivered to the construction orbit.

Of possible interest is the storable/SEP combination that approximates the weight of the cryogenic propulsion system. The trip time of such a system is approximately half of what an all-SEP system would be. In addition, the smaller SEP with correspondingly smaller solar array requirements could prove to be more economically attractive because it would be closer to required GEO levels.

The hybrid propulsion concept could be attractive if utilization of existing chemical stages (or stages sized to other missions) becomes important.

5.2.4.5 References

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2. Masek, T. D., et al., *Advanced Electrostatic Ion Thruster for Space Propulsion*, Hughes Research Labs, Malibu, CA., April 1978 (NASA CR-159406).

3. Wilbur, P. J., *Scaling Relationships for Hg and Gaseous Propellant Ion Thrusters*, 13th International Electric Propulsion Conference, San Diego, CA., April 25-27, 1978 (Paper 78-667).

4. Byers, David C. et al., *Primary Electric Propulsion for Future Space Missions*, loc cit, Ref. 1, AIAA 79-0881.

5. Robey, D. H., *Electric Cargo Orbital Transfer Vehicle Parametric Sizing Study*, Internal Memo, IL 590-200-79-018, December 22, 1978, Satellite Systems Division, Rockwell International

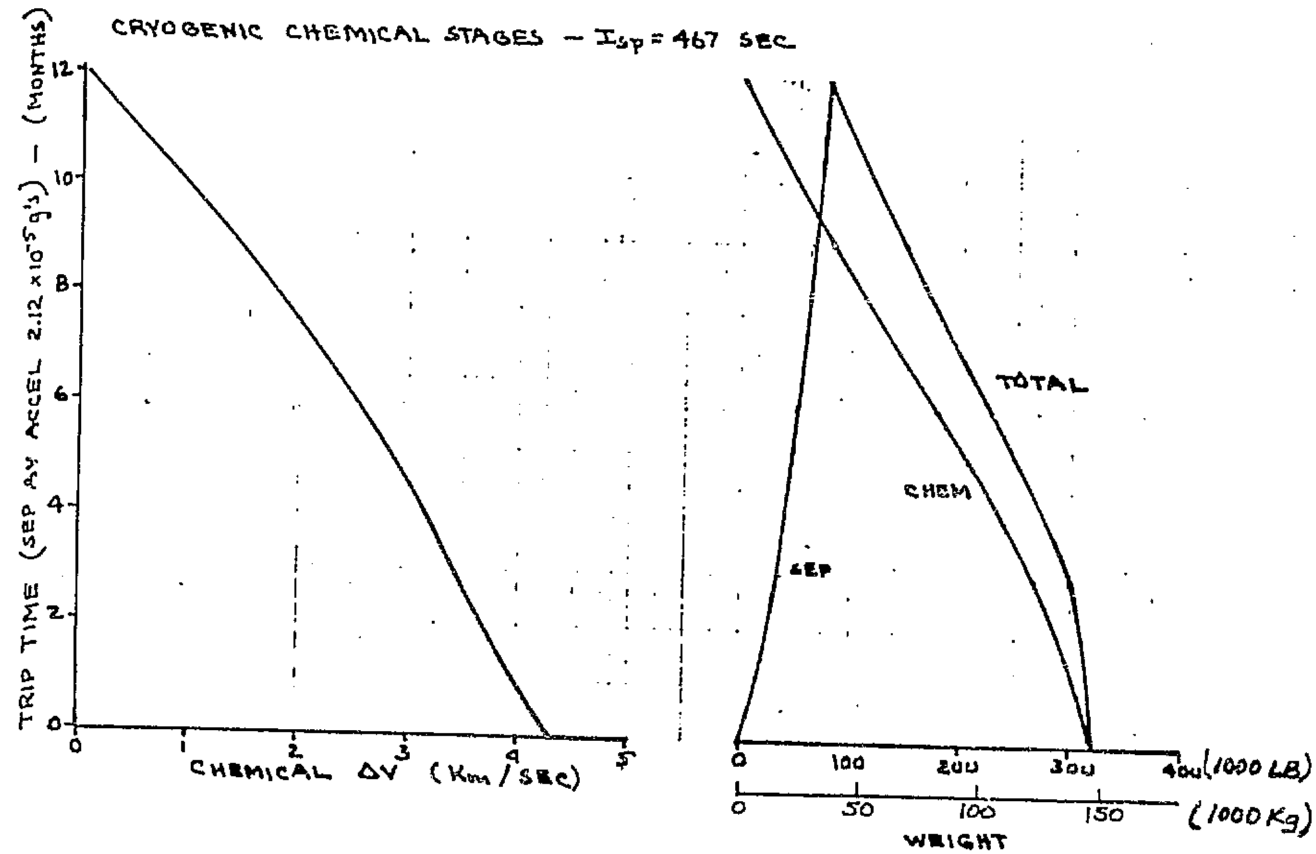


Figure 5.2-59. Hybrid SEP plus Chemical Propulsion Module Characteristics (Cryogenic)

Satellite Systems Division
Space Systems Group

Rockwell International

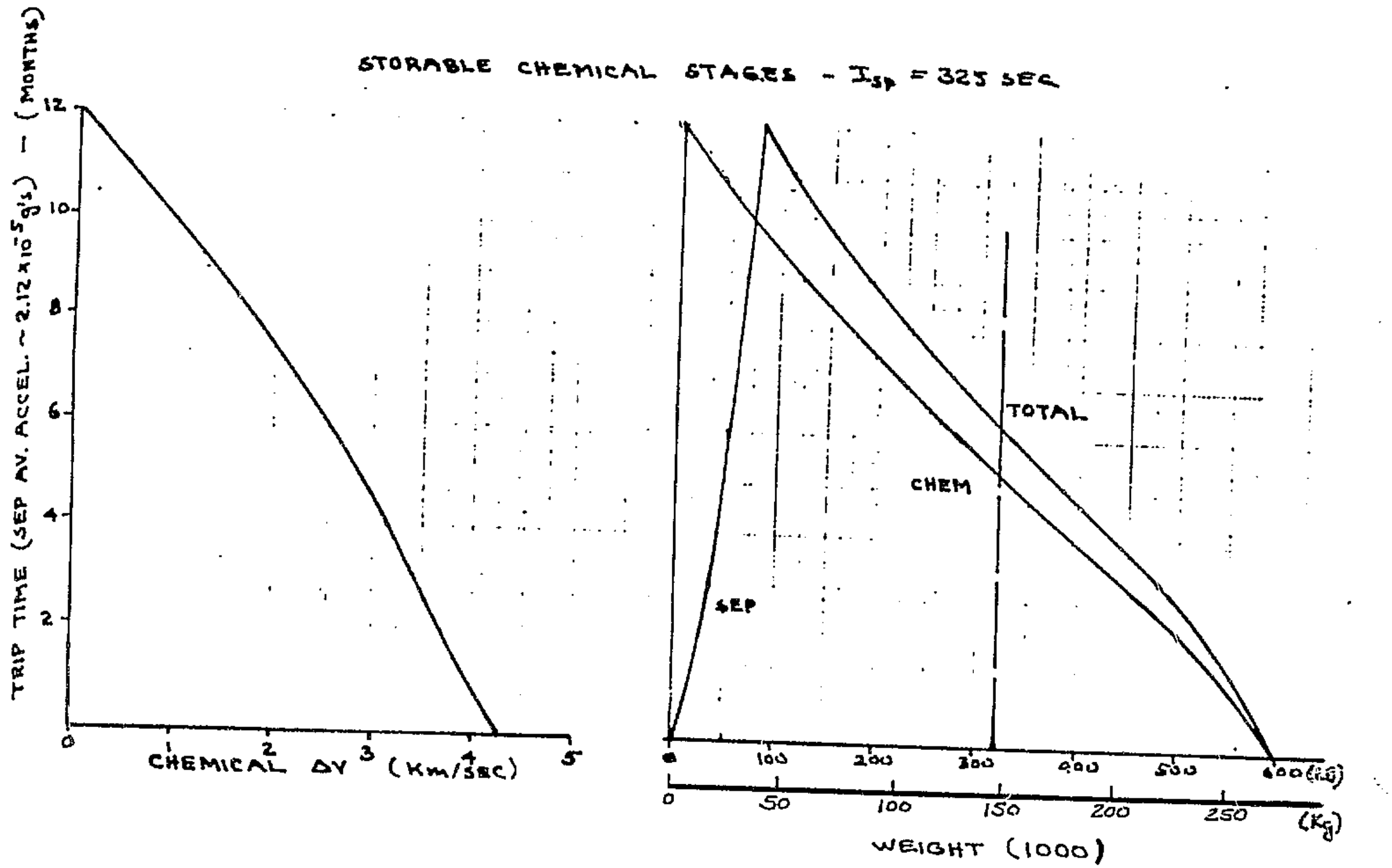


Figure 5.2-60. Hybrid SEP plus Chemical Propulsion Module Characteristics (Storable)

5.2.5 Orbit Transfer Summary

The preceding discussions have individually developed the main sizing features and governing parameters along with Shuttle delivery requirements for four basic types of propulsion. The purpose was to explore their suitability for use in the orbit transfer of large area space systems. Figure 5.2-61 summarizes the principal comparative factors. Based on these results, an advanced cryo stage appears to offer the most promise as a general-purpose orbit transfer vehicle for use with large area space systems. Properly designed project systems can easily function with moderate thrust loads imposed by cryo stages. Use of multiple nozzles, judicious staging and/or possible throttling can keep the peak thrust loads down. Control frequencies for thrust steering can be adequately separated from structural bending frequencies for proper stability without excessive ΔV penalties.

The higher thrust loads necessary with the IUS solid motor introduce complications into the design and construction of the project system. Also, its lower I_{sp} performance, like that of the advanced storable, requires significantly more propulsion system weight than the cryo and, thus, many more Shuttle flights for delivery.

Solar electric propulsion requires very large (and costly) solar arrays, resulting in excess electrical power over that neeeded for GEO mission operations. It also requires long trip times—six months or longer—which can lead to significant investment cost increases for space projects like the advanced communications platform treated here. The extra trip time over a cryo OTV concept would defer the revenue generating phase of its operations. With the 1300-plus transponder capacity of the example platform, this revenue deferral could represent tens and possibly hundreds of millions of dollars. However, if solar electric propulsion were to become available through continued development in support of other programs such as SPS, particularly with advances in technology and performance to improve its cost effectiveness, it could be a serious candidate for orbit transfer of large area space systems.

The advanced cryo concept still appears to offer the most promise. The need for a manned OTV capability seems inevitable. Because of its wide performance margin over other types of propulsion, and its trip time compatibility with manned OTV requirements, the cryo concept is felt to be the strongest candidate to fill the manned OTV need. Actually, it can do both jobs, the transfer of large area space systems and the transport of manned OTV's to GEO and other high-energy missions. Thus, it is the logical choice of the propulsion concepts considered. Other propulsion concepts can probably be developed, which would be less costly on any one given project, but as a general-purpose OTV serving both manned and unmanned applications, the advanced cryo concept offers the most in terms of performance and versatility.

In addition to the propulsion concept assessments in the preceding discussion, different propulsion delivery techniques were briefly investigated. Figure 5.2-62 summarizes the comparative factors of the four basic delivery concepts investigated. The platform flydown approach is a clear favorite. It offers major performance advantages over the other approaches (fewest Shuttle flights for propulsion delivery) and permits manned participation in the propulsion installation process.

	LO-THRUST CHEMICAL			
	CRYO	STORABLE	IUS	SEP
$(T/W)_{max}$	0.2	0.2	0.9	10^{-4}
NUMBER OF MODULES	5	10	24	N/A
NUMBER OF STAGES	2	3	12	N/A
NUMBER OF LAUNCHES (DEDICATED TO PROPULSION)	6	10	12	3 - 4
IMPACT ON SOLAR ARRAY	HINGING REQ'D			>5 x AREA
IMPACT ON ANTENNAS	PARTIAL DEPLOYMENT		NO DEPLOY	FULL DEPLOY

Figure 5.2-61. Propulsion Comparison Summary

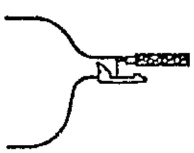
	DELIVERED PROPULSION WT, LB 300 NMI CONST ORBIT	DELIVERED PROPULSION WT, LB 250 NMI CONST ORBIT	COMMENT
 DIRECT SHUTTLE DELIVERY TO CONST ORBIT	OFF LOAD TO 50,500 LB 6.1 MODULES REQ'D	OFF LOAD TO 56,900 LB 5.4 MODULES REQ'D	LOW DELIVERY PERFORMANCE
 "SELF" DELIVERY	62,000 LB 5 MODULES REQ'D	62,800 LB 5 MODULES REQ'D	REQUIRES GUIDANCE AND RCS EA MODULE
 TELEOP DELIVERY	63.500 LB 5 MODULES + 3.5 TELEOP	63.500 LB 5 MODULES + 2.5 TELEOP	REQUIRES SPACE BASED TELEOP FARM AND/OR SPACE REFUEL
 PLATFORM "FLY DOWN" TO SHUTTLE ORBIT	63,500 LB 5 MODULES + 1.1 TELEOP	63,500 LB 5 MODULES + 0.6 TELEOP	GOOD LOGISTICS EFFICIENCY MAN-TENDED MODULE INSTALLATION

Figure 5.2-62. Propulsion Delivery Techniques Comparison

Direct Shuttle delivery to the construction orbit imposes performance penalties due to the sharp drop in payload capability above 370 km (200 nmi). Propulsion self-delivery requires greater stage complexity and costs in order to individually perform the precision rendezvous and docking with the space platform in its construction orbit. Teleoperator approaches require either multiple teleoperators, grouped in "farms," or in-space refueling of a single teleoperator along with additional complexities.

Thus, the platform flydown approach is identified as the prime candidate for propulsion delivery. Other project systems constructed at different orbit altitudes could require further consideration of some of the other delivery approaches, but the flydown concept appears to be a serious contender for all missions where the construction orbit is substantially above the maximum payload altitude of the Shuttle [370 km (200 nmi].

APPENDIX

CONSTRUCTION FIXTURE DESIGN TRADES

A.1 INTRODUCTION AND SUMMARY

The principal issues associated with the design of the primary construction fixture for the construction and assembly of large area spacecraft will be discussed here. The appendix will also discuss the generic issues associated with all space construction projects, and illustrate the design concepts developed for the three particular construction projects: (1) the Satellite Power System (SPS) Test Article, (2) the Space-Fabricated Communications Platform, and (3) the Erectable Communications Platform. The design of these fixtures will be based on construction being accomplished from the Shuttle orbiter as the construction facility. The influence of construction from a space base will be addressed as a separate issue in another report—Potentials and Implications for a Space Construction Facility.

Five principal construction fixture requirements have been identified and described, and are listed below:

1. Support the construction project during fabrication and assembly
2. Provide translation capability
3. Provide orbiter revisit capability
4. EVA provisions
5. Packable for orbit transport

Three construction fixture design issues have been identified and their influence on the design and operation of the fixture has been described; these issues are:

1. Orientation of the construction fixture with the orbiter
2. Construction and assembly effort in the vicinity of the fixture
3. Single versus multiple beam builders.

The three construction fixture baseline configuration concepts have been described for each of the three construction projects. The principal construction fixture requirements and the trades performed to select the baseline fixtures have also been described. The final construction fixture design will be the result of an integrated, overall, construction process and is part of the construction analysis iterative process as illustrated in Figure A-1.

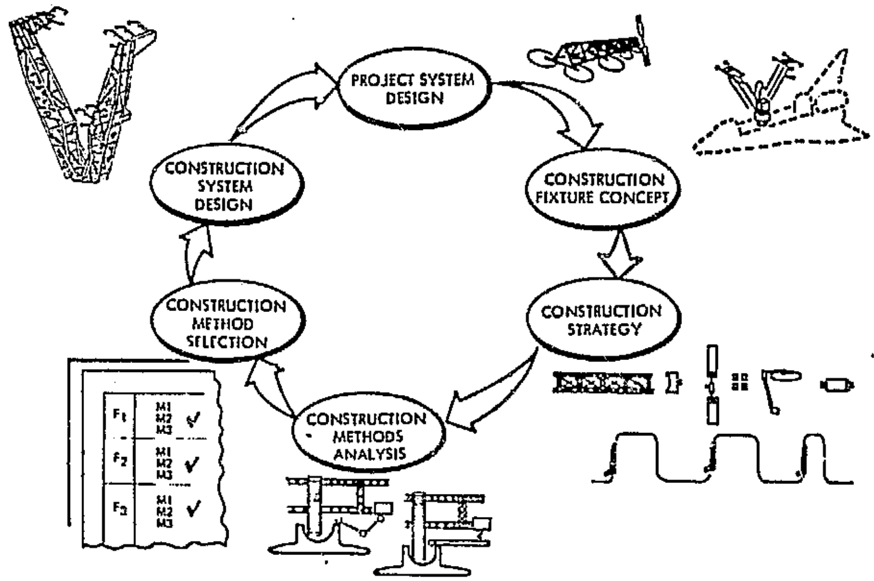


Figure A-1. Construction Analysis Iterative Process

A.2 CONSTRUCTION FIXTURE REQUIREMENTS

There are certain fundamental requirements that must be considered for all primary construction fixtures used for large area spacecraft/platform construction and assembly. Unique construction support equipment may be necessary to augment the primary construction fixture in order to accomplish particular tasks. These unique equipment items are dependent on the particular method of construction or assembly associated with individual construction projects. Therefore, these items are discussed in the Data Base Methods Description.

Figure A-2 lists the fundamental construction fixture requirements. The principal function of the construction fixture is that of a master tool. The fixture provides the precise location of all of the primary structural members and must secure the members in their correct location during the assembly process. In this role the construction fixture also acts as a mounting base to which the construction support equipment is attached.

The primary construction fixture should also have the capability to translate the construction project through the fixture. This capability permits flexibility in the selection of construction methods. This issue will be discussed in the following section on Fixture Issues. The translation capability must allow project translation in various stages of completion. Consequently, the construction process must be determined in order to develop the approximate fixture configuration.

With the Shuttle Orbiter as the construction base the capability for orbiter revisit is required. Stabilization of the fixture/project to permit orbiter revisit maneuvers and attachment may be required, but is dependent on the particular construction project and its configuration at the revisit period. In addition to the physical attaching provisions the utilities services interfaces for electrical power, data and control must also be provided.

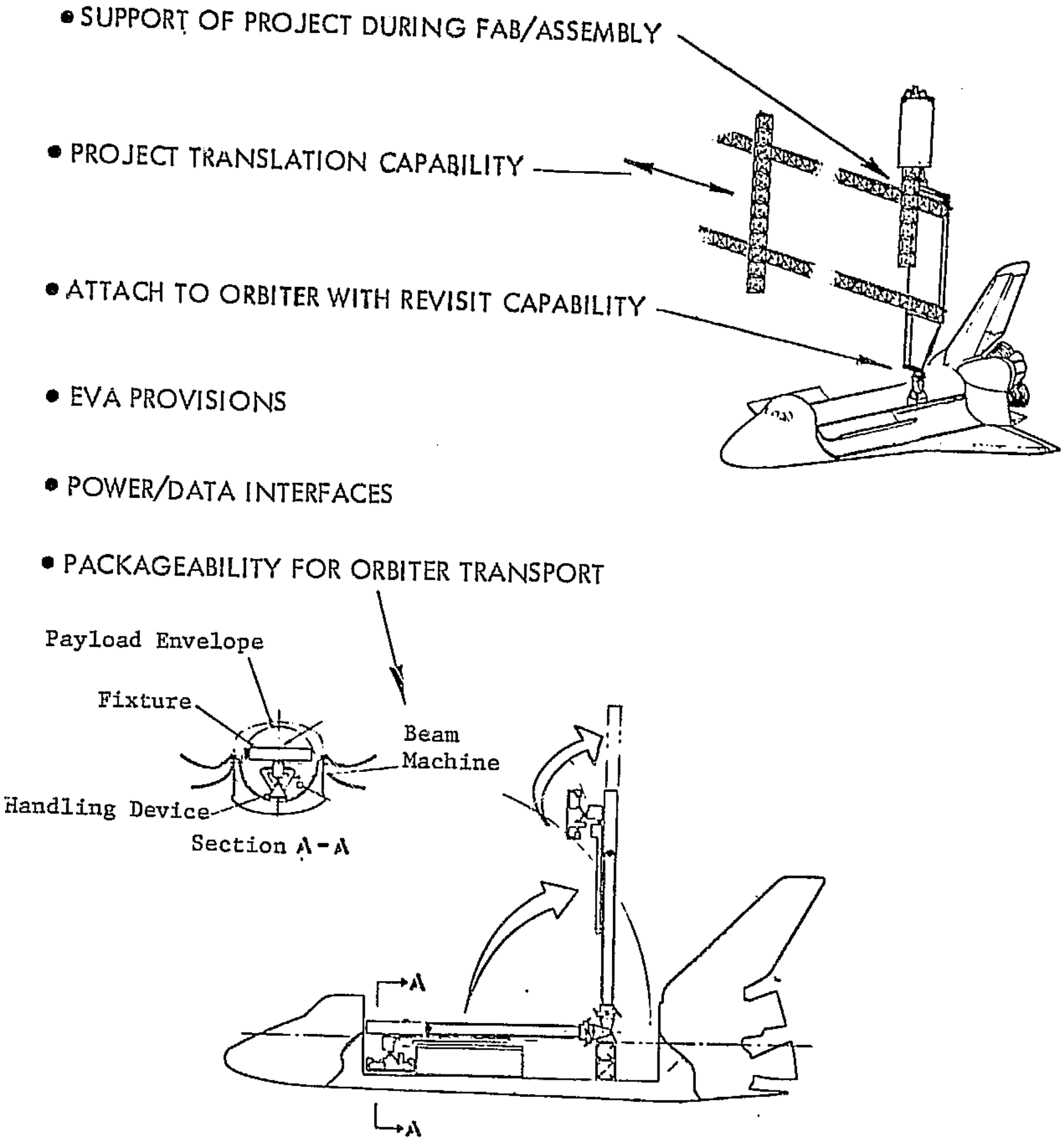


Figure A-2. Space Construction Fixture Requirements

All of the construction fixtures must have provisions for EVA contingency operations. The contingency provisions are of particular importance for those automatic and semi-automatic construction methods. EVA provisions, of course, are provided for construction operations that utilize the EVA mode.

Interface provisions to accept electrical power for the operation of the various construction device and illumination is necessary. Feedback and control circuit interfaces is also necessary in order to operate, control, and monitor the construction operations.

Probably the most demanding design requirement is that of packaging the construction fixture to fit within the orbiter payload bay. This requirement applies for both construction from the orbiter or construction from a space base. This requirement may demand the fixture to be folded; portions removed and assembled in orbit, or sections attached only as required for a particular assembly sequence.

A.3 CONSTRUCTION FIXTURE ISSUES

During the course of this study, certain operational issues were addressed that influence the design concept arrangement of the primary construction fixture. These issues concern, 1) the orientation of the construction/assembly fixture in relationship to the orbiter, 2) the location of the construction/assembly work station in the vicinity of the fixture, and 3) serial or parallel operations. A discussion of each of these issues follows.

Orientation of Fixture

Certain orbiter operational constraints exist which must be addressed when the orbiter is utilized as the construction facility. These constraints influence the location of the construction fixture and influence the direction of construction for the project being assembled. The principal issues addressed in this orientation trade study are 1) locate the fixture and project to provide clear access to the payload bay, 2) locate the construction operation within the reach envelope of the RMS, 3) locate for maximum direct visibility from the orbiter cabin, 4) locate to clear the orbiter for project translation during construction, and 5) locate the fixture and the project so as to minimize orbiter radiator shadowing.

In order to deliver the construction project components to their proper location for assembly the payload bay must be clear and the RMS must be free to transport the components to their proper location. Consequently, the construction fixture interface with the orbiter is limited to either end of the payload bay. The location in the aft end of the payload bay may be more desirable because there is no possible interference with the RMS operation with, however, some sacrifice in reach.

Direct visibility from the orbiter crew cabin aft and overhead windows is desirable in order to verify general clearances during the fab and assembly operations. However, detail installations will probably require T.V. augmentation to verify the particular operations. Fixtures located in the aft end of the orbiter are more capable of being viewed from the aft crew cabin windows than fixtures located in the forward end of the payload bay. Fixtures located in the forward end of the payload bay tend to obscure the vision of operational activities as well as payload bay activities. Payload bay visibility is necessary for the RMS operation of transporting project components or performing subtasks within the payload bay.

The capability to translate the construction project through the fixture in either direction has been determined to be a desirable feature. This capability allows greater flexibility in the determination of the construction sequence. It also permits flexibility in the payload manifest, thus, providing the best packaging density. It also allows for unscheduled maintenance activities. Probably the most important feature of this capability is that it permits the construction operations to be performed in the immediate vicinity of the construction fixture and orbiter. This concept will be discussed in greater detail in the Location of the Construction/Assembly Effort Section.

The translation capability, however, incurs complexities in the design and operation of the construction fixture. For the space-fabricated type of structures, such as represented by the SPS test article or communications platform tri-beam configuration, the structure supporting device must have the capability to "step over" the cross beams during the translation operation. The location of the project must allow the translation motion to clear the orbiter for all of the project configurations resulting from the assembly sequence. This requirement may dictate the size of the fixture depending on the distance the construction project must be located away from the orbiter to permit clearance for translation.

Translation of erectable type structures as represented by the erectable communications platform is different than that of the space-fabricated structural arrangements. Since the erectable structure does not have continuous longitudinal beams as the space-fabricated structure does, the translation must be accomplished in steps rather than a continuous motion. The steps would be in increments of the pentahedral base dimension. This concept requires a translation device and a device to hold the structure while the translation device releases the structure and returns to its starting position.

Heat rejection from the orbiter is a critical function and, therefore, precautions must be taken to obtain the maximum heat rejection capability of the orbiter system. This concern requires that during construction from the orbiter, the orbiter radiators are not shadowed by the construction fixture or the construction project. This shadowing will reduce the radiation capability and may require the orbiter to use the water boiling auxiliary heat rejection system.

<u>Location of the Construction/Assembly Work Station</u>

From this study the desirability to perform the construction operations in the immediate vicinity of the construction fixture rather than performing operations at some distance from the fixture/orbiter was determined. By keeping the work stations close-in to the area to be worked, the number and complexity of construction support devices can be minimized; the parts storage area (the orbiter payload bay) is close; and direct visibility from the orbiter crew cabin is possible. Lighting of the area and the installation of auxiliary T.V. viewing is more easily accomplished in this location. The T.V. cameras located on the forward and aft bulkheads of the orbiter payload bay can be utilized when the construction operations remain close to the construction fixture. The safety of operations in this location is enhanced due to the proximity for assistance if required and because of the more direct monitoring of the activities.

Serial or Parallel Operations

Performing construction operations, particularly the fabrication and assembly of the primary structure, can be accomplished in a serial mode or in a parallel mode. The selected method has a significant effect on the construction fixture configuration.

For space fabrication and assembly operations the serial mode would utilize one beam builder while the parallel mode would utilize multiple beam builders simultaneously. For instance, the longitudinal members of the SPS Test Article could be fabricated by utilizing two beam builders operating simultaneously. This operation method would eliminate the device required to move a single beam builder from one position to the other. A third beam builder could generate the transverse members. If all three were operated simultaneously, the total structure could be completed during the initial translation of the platform structure.

Erectable structure assembly can utilize a subassembly mode in a parallel operation. Subassembly stations can be assembling a cluster of struts and unions while the assembly of a similar cluster is being assembled into the basic structure. The additional subassembly station facility minimizes the number of individual strut/union members supporting and locating devices necessary on the construction fixture to locate and hold each member in its proper relationship.

A trade study considering such issues as cost, fabrication and assembly time, complexity of the construction fixture, etc., is required in order to select a preferred serial or parallel construction method.

A.4 CONSTRUCTION PROJECTS

As examples of the implications of the construction fixture issues a discussion of the baseline construction fixtures developed for each of the study construction projects will follow.

SPS Test Article

The configuration of this construction project (Figure A-3) consist space-fabricated ladder type structure 215m long and 20m wide to which ar attached 25 solar array blankets. RCS modules are installed on the ends two cross beams. Solar electric propulsion modules provide the orbit transfer propulsion.

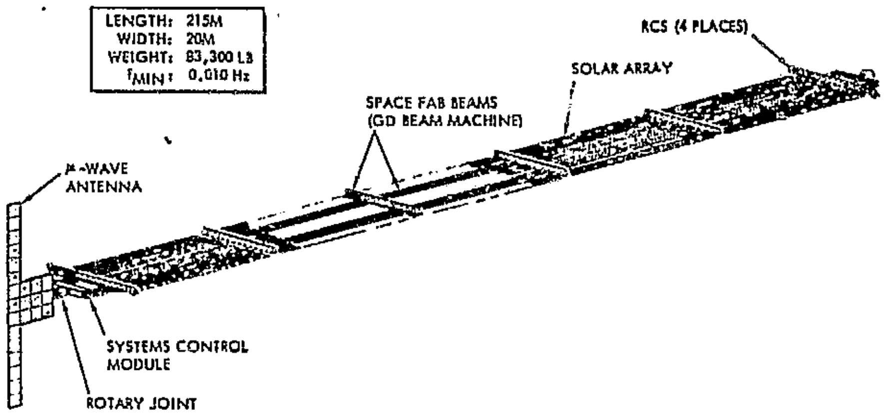


Figure A-3. SPS Test Article General Arrangement

In conjunction with the construction fixture design concept, a baseline construction sequence was developed (Figure A-4). With this sequence and the construction project configuration/definition, a baseline construction fixture was identified (Figure A-5). The issue concerning the location of the fixture in relationship to the Shuttle orbiter was then addressed. Figure A-6 summarizes the location trade and six principal evaluation factors. The construction sequence requires a translation ability which drives the location in relationship to the orbiter and to the fixture configuration. Retaining clear access to the payload bay and providing minimum shadowing of the orbiter radiators also influence the location of the fixture and construction project orientation.

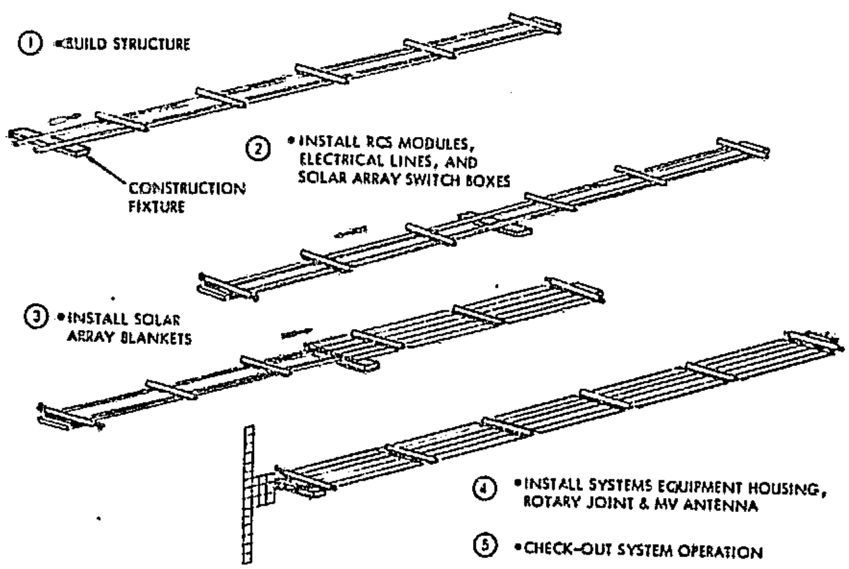


Figure A-4. Construction Sequence

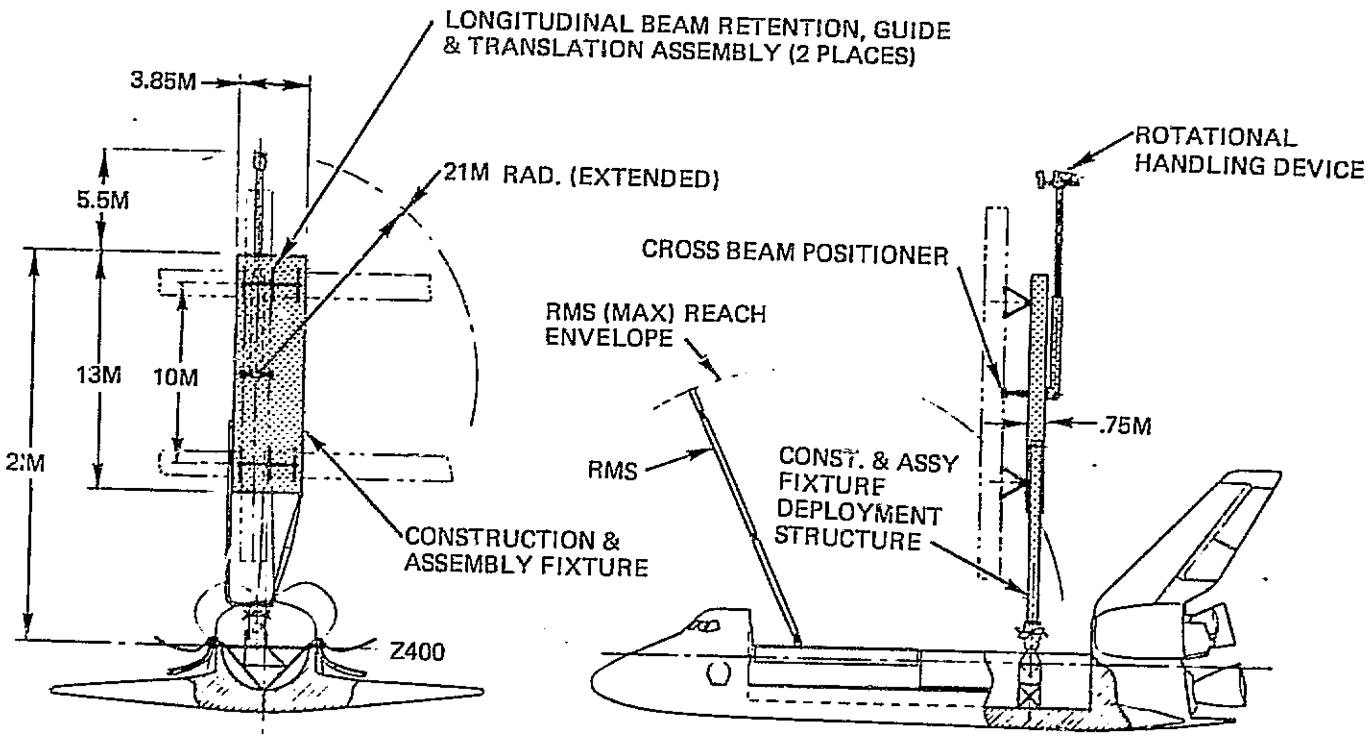
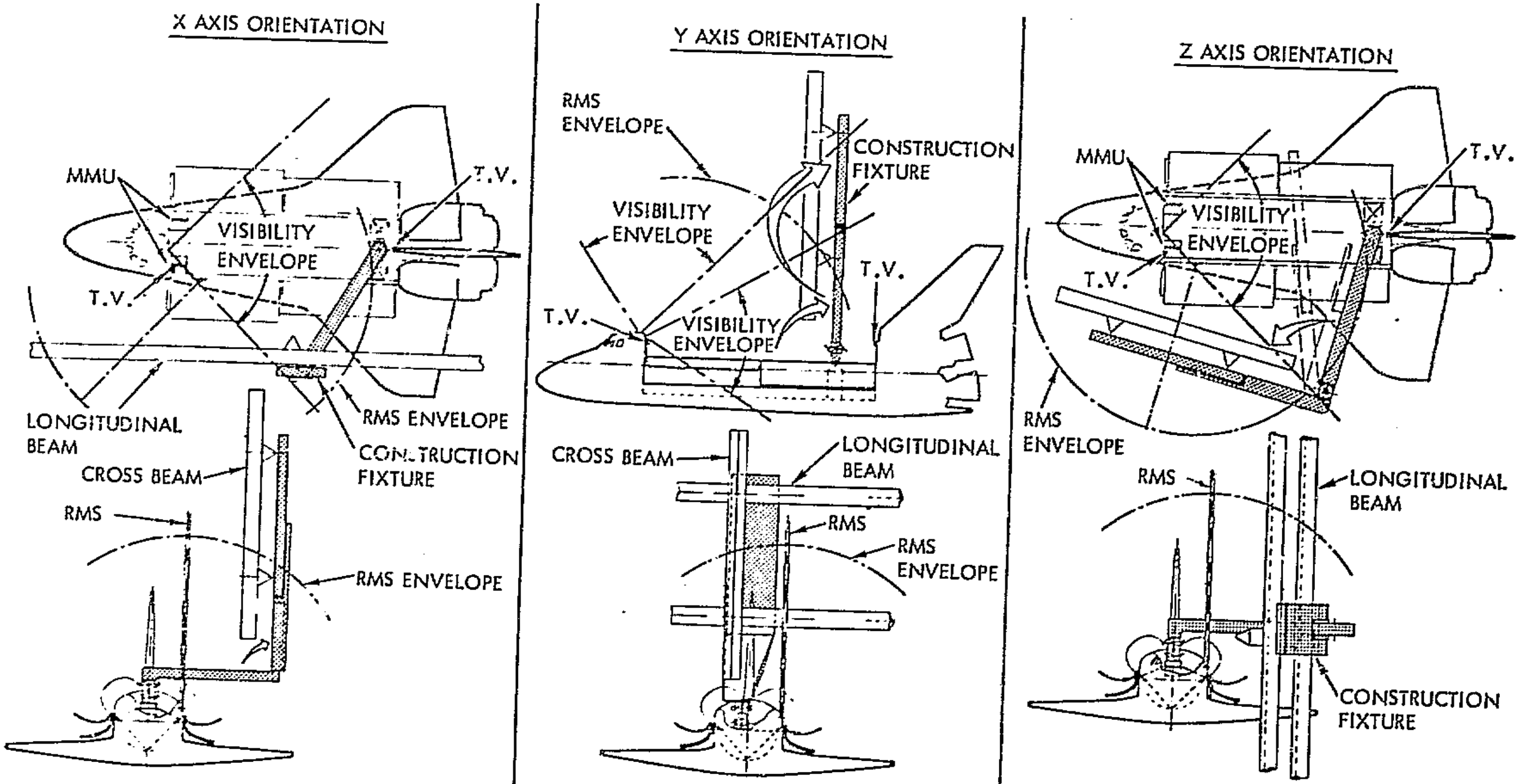


Figure A-5. SPS Test Article Construction Fixture



X Axis Orientation

- CLEAR ACCESS PAYLOAD BAY
- MIN. ORBITER RADIATOR SHADOWING
- LIMITED RMS REACH
- DIRECT VISIBILITY - ACCEPTABLE
- TRANSLATION CAPABILITY
- LONG BOOM FIXTURE SUPPORT

Y Axis Orientation

- CLEAR ACCESS PAYLOAD BAY
- MIN. ORBITER RADIATOR SHADOWING
- LIMITED RMS REACH
- DIRECT VISIBILITY - ACCEPTABLE
- TRANSLATION CAPABILITY
- DIRECT FIXTURE ATTACH

Z Axis Orientation

- CLEAR ACCESS PAYLOAD BAY
- MIN. ORBITER RADIATOR SHADOWING
- WITHIN RMS REACH ENVELOPE
- DIRECT VISIBILITY - POOR
- TRANSLATION CAPABILITY
- LONG BOOM FIXTURE SUPPORT

Figure A-6. Construction Fixture Concepts Trade

Maintaining the construction activities within the RMS envelope is desirable because it permits transport of items from the payload bay to the construction site without the necessity to provide a special piece of equipment to perform this function. It also permits the option to use a cherry picker on the end of the RMS as an assembly mode. Visibility from the crew cabin is desirable as an overall monitoring method, but can be augmented with T.V. coverage. The "Y" axis orientation was selected for the baseline arrangement because of the simplicity of the construction fixture and because of the direct visibility from the crew cabin. However, the "Z" axis orientation permits the construction operations to be performed within the reach envelope of the orbiter RMS which cannot be achieved with the "Y" axis orientation. This capability permits the construction operation via a "cherry picker" attached to the RMS a feasible method. Consequently, during the development of alternate methods for performing the various construction functions the "Z" axis orientation will also be considered.

The translation capability incorporated in the construction fixture permits the operations to always be performed in the vicinity of the fixture. This arrangement satisfies the second of the construction fixture issues.

A single beam builder was utilized for the construction effort of this project. As previously mentioned, the trade between single or multiple beam builders requires a total integrated concept for evaluation.

Preliminary analysis has indicated that construction in a free drift (gravity gradient) mode is feasible and, therefore, no attitude control during construction would be necessary if the analysis is correct. Attitude control for the rendezvous and docking maneuver has not been established. If attitude control is required for rendezvous and docking, the construction fixture would then be capable of achieving this requirement.

Advanced Communications Platform - Space-Fabricated Concept

The configuration of the space-fabricated communications platform (Figure A-7) consists of a tri-beam structure to which sixteen various sized antennas are mounted. A deployable type solar array is mounted on one end of the structure while a thrust structure accommodating five low thrust chemical propulsion modules is mounted on the opposite end of the structure.

The construction sequence established for this concept is illustrated in Figure A-8. The development of the basic construction fixture (Figure A-9) required a number of trade studies. Figure A-10 illustrates the essence of the trade performed to address the issue of the construction project translation.

The translation capability involves a fixture concept that will provide support to the construction project during translation and be attached to the construction facility, i.e., the orbiter.

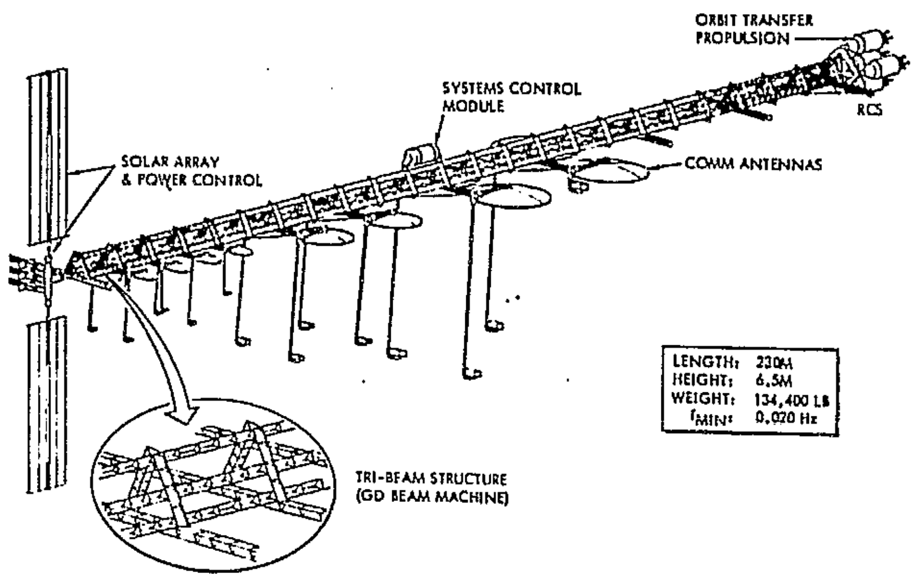


Figure A-7. Communications Platform Space Fabricated Structure Concept

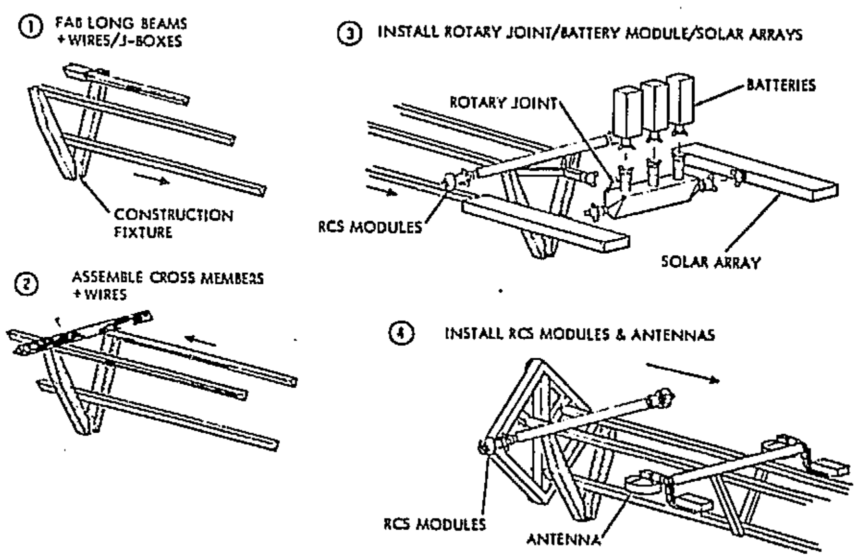


Figure A-8. Construction Sequence

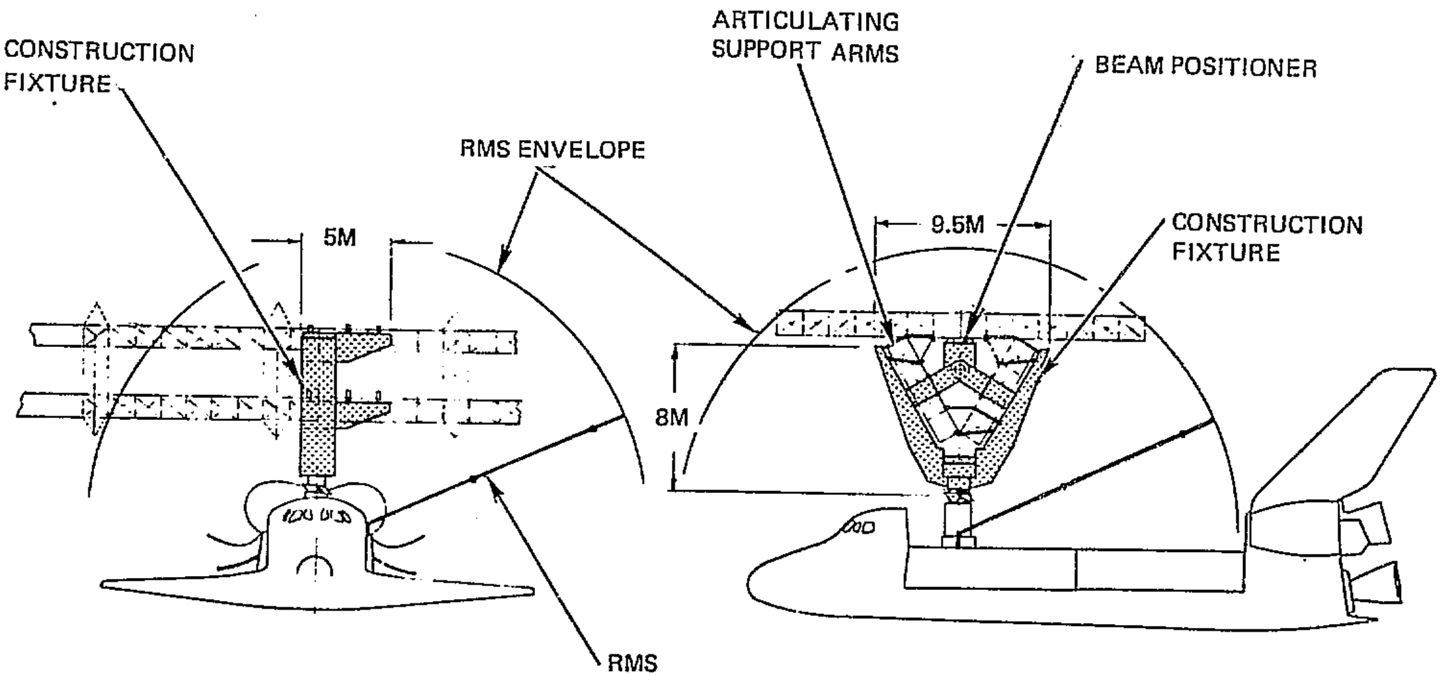


Figure A-9. Space Fabricated Tri-Beam Construction Fixture

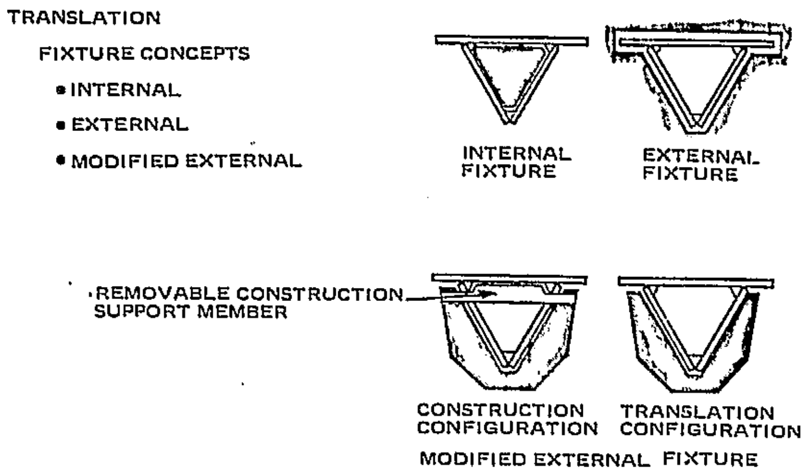


Figure A-10. Translation Trade

The internal fixture concept illustrated in Figure A-10, is the minimum sized fixture that could be utilized to support a construction project. However, the support of the fixture to the facility requires a complex arrangement that would permit the passage of a transverse member through the support structure during the translation activity.

The external fixture, however, requires no special arrangement to permit translation to occur. However, the fixture must completely surround the construction project thus making a very large unit that would be difficult to transport and assemble on orbit.

The modified external fixture concept utilizes features of the other two concepts. The external portion of the fixture permits translation of the construction project by utilizing articulating project support arms that permit the transverse members to pass through the fixture. A removable cross member of the fixture provides support for the beam builder and other structure construction support devices. After the structure is complete the cross member is removed thus permitting the project translation to occur.

The modified external fixture concept was selected for the construction of the space-fabricated tri-beam advanced communications construction project.

The location of the fixture in relationship to the orbiter is illustrated in Figure A-11. Except for the direct visibility blockage in this location, the other location issues are acceptable such as the RMS reach envelope, clear payload bay, etc. Having achieved the payload translation capability, the construction activity will be accomplished in the vicinity of the fixture.

Figure A-12 summarizes the tradeoff concepts that were developed to utilize one beam builder to perform the construction operation. Three concept arrangements are illustrated for fabricating the longitudinal beams and three for the fabrication of the cross beams.

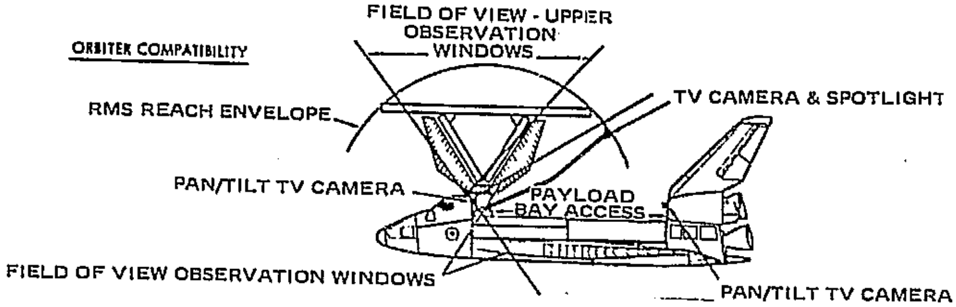


Figure A-11. Fixture/Orbiter Orientation

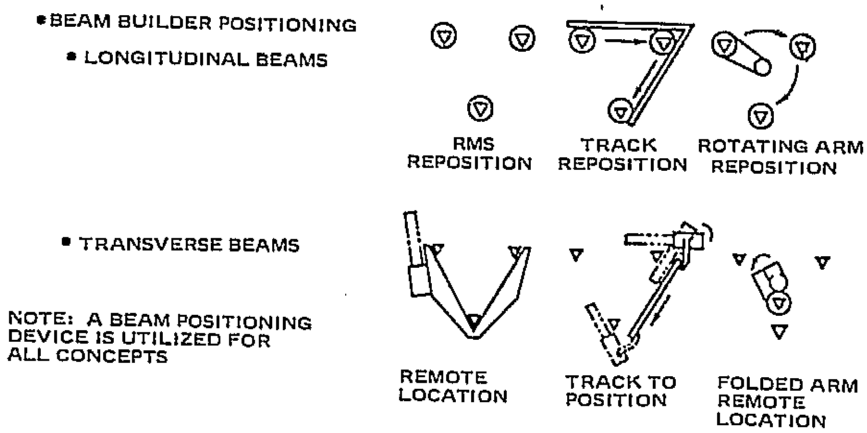


Figure A-12. Beam Builder Operations Trade

The beam builder, in order to fabricate the longitudinal members, must fabricate the beams at the fixture location that secures the members to the fixture in their proper final location. The three concepts shown illustrate three methods of repositioning the beam builder. The RMS can place it on the fixture at the proper location for each longitudinal beam; the beam builder can be mounted on a track that will guide the beam builder to the proper position; and the beam builder can be mounted to a rotating arm that positions the beam builder in the proper location.

Before selecting a concept for positioning the beam builder to fabricate the longitudinal beams, a similar option trade must be done for the beam builder location to fabricate the transverse beams. Three concepts are illustrated. The remote location places the beam builder on the side of the fixture. The completed transverse members are then transported to their proper assembly location. The track concept guides the beam builder as illustrated. The transverse beams in this concept are fabricated in the location where they will be used. The folded arm concept utilizes the rotating arm concept for fabricating longitudinal members. A hinge joint in the rotating arm allows the beam builder to locate between the longitudinal members. The transverse members are generated parallel to the longitudinal beams and are then transported to their proper assembly position. The folded arm concept is only feasible when the tri-beam section is sufficiently large to permit the beam builder to clear the longitudinal members.

The selected arrangement must be an integrated arrangement that is advantageous for the fabrication of both members. Figure A-13 illustrates an integrated fixture concept that was developed for the construction of a tri-beam type structure. The final fixture concept for this project can only be determined after all of the construction functions have been investigated and an integrated construction process has been developed.

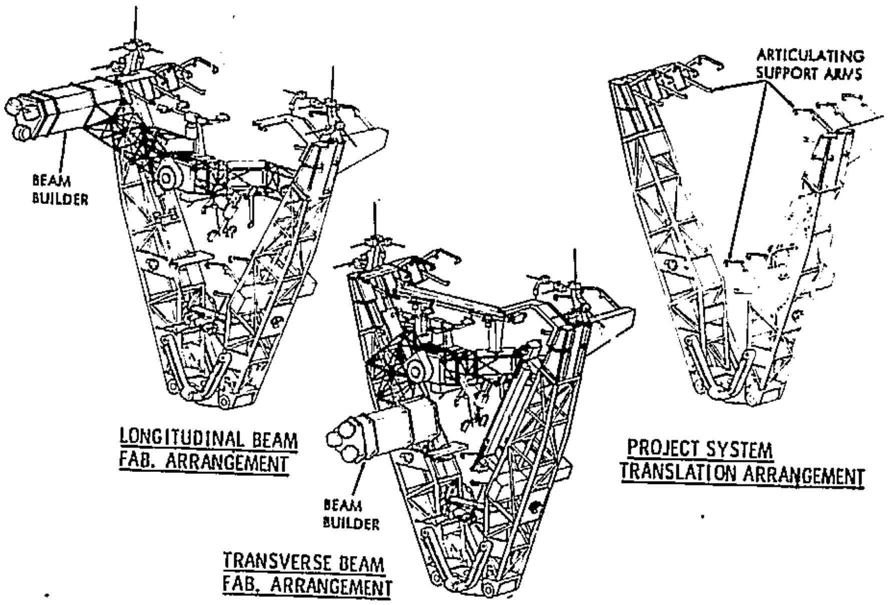


Figure A-13. Integrated Construction Fixture Concept

Advanced Communications Platform - Erectable Structure

The erectable structure concept of the communications platform (Figure A-14) is similar to the space-fabricated concept. The structure consists of tapered tubes joined to form pentahedral units. These units are joined to form the basic structure and "outrigger" type supports are added to accommodate the RCS modules. A similar type of outrigger structure provides the support for the solar array panels and the low thrust chemical propulsion pods.

The construction sequence is illustrated in Figure A-15. As previously mentioned, the translation capability of this type of structure must be accomplished in steps because there are no continuous longitudinal members. This unique condition significantly influences the basic construction fixture.

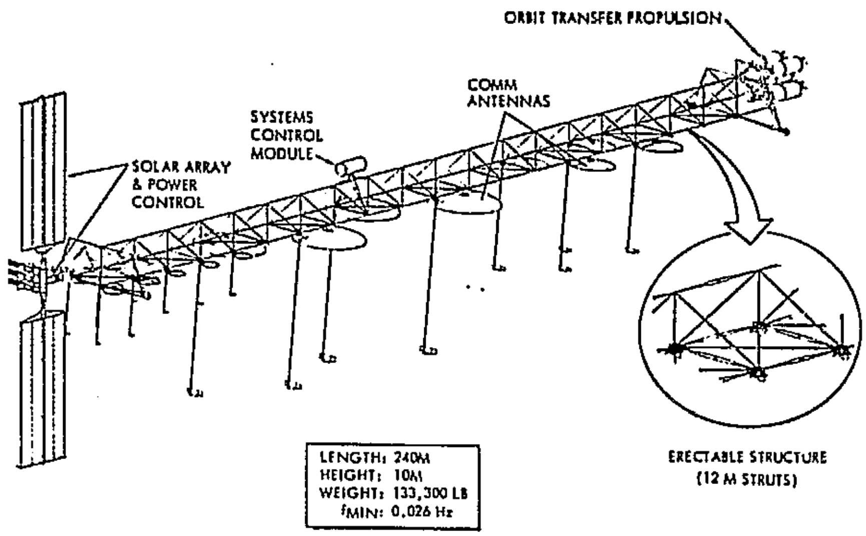


Figure A-14. Communications Platform
Erectable Structure Configuration

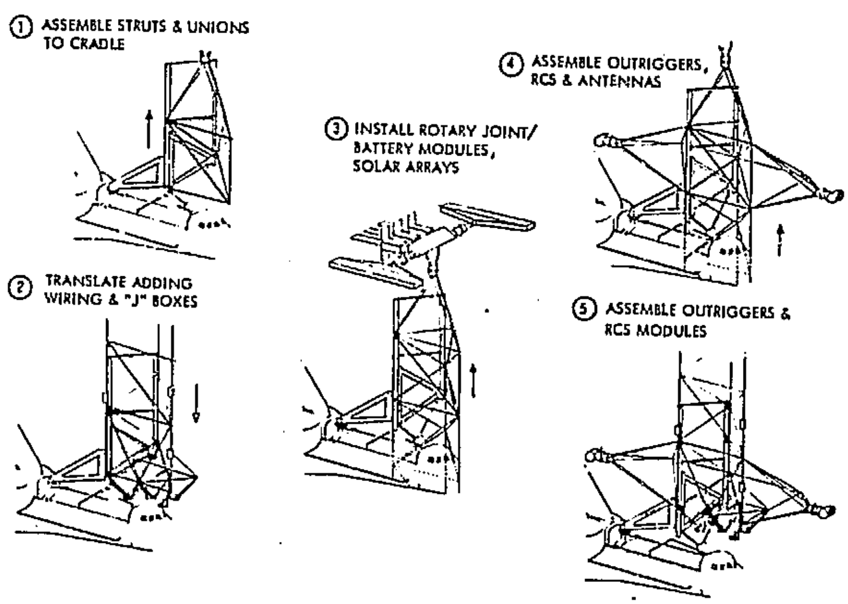


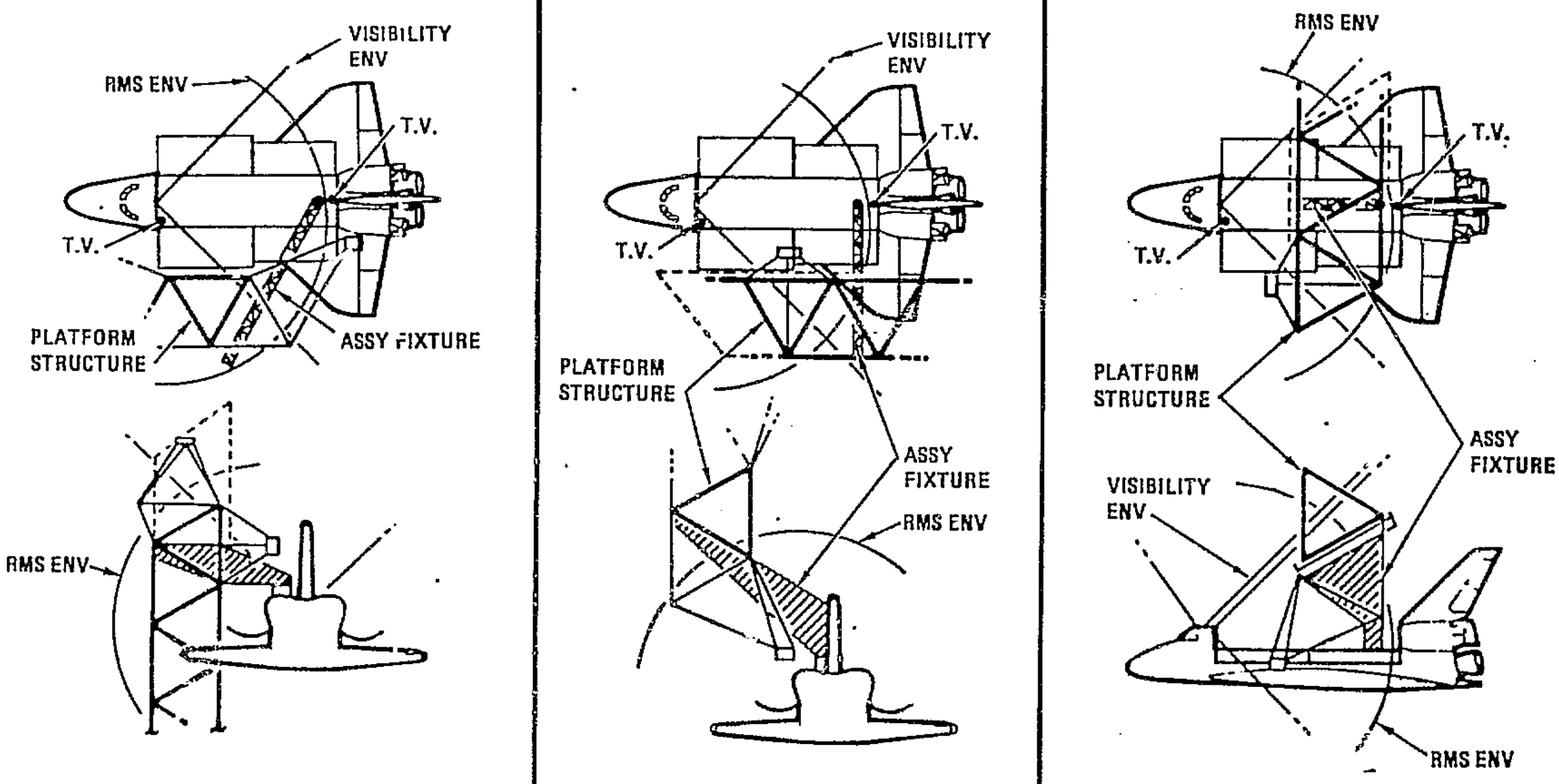
Figure A-15. Construction Sequence

A trade study concerning the location of the fixture/project in relationship to the orbiter was performed and is summarized in Figure A-16. Five evaluation factors were used to select a preferred baseline orientation. The dominant evaluation factors were the translation capability and the RMS reach envelope. The "Z" axis orientation is the only one of the three concepts that provides the translation capability and is mainly within the reach envelope of the RMS. Direct visibility from the crew cabin for the Z axis orientation is very marginal and the visibility of construction activities will most likely need to be accomplished with the use of T.V. cameras. The "Y" axis orientation tends to block the orbiter payload bay access. The construction project in the "Y" axis orientation creates greater orbit radiator shadowing than the construction fixture support boom of the "Z" and "X" axis orientation arrangements.

The "Z" axis orientation arrangement was selected as the baseline construction fixture arrangement. This concept permits the construction operations to be performed within the RMS reach envelope, thus minimizing the complexity of the fixture.

The upper portion of Figure A-17 illustrates the baseline fixture configuration that was developed for the construction of the erectable antenna platform concept. The fixture consists of a 2 guide rail arrangement that permits a platform translation cradle to translate the construction project in steps. However, during the analysis of the construction operations alternative methods, the assembly of the thrust structure and the RCS module support structure was interfering with the outboard guide rail. In order to eliminate this interference and to reduce the extreme reach required of the RMS, the revised fixture baseline configuration illustrated in the lower portion of Figure A-17 was developed. The revised fixture utilizes a single guide rail and adds a rotation of the platform translation cradle to bring the platform within the RMS reach envelope for all sections of the construction project. However, the final fixture configuration can only be determined after an integrated construction procedure and operations has been determined.

Z AXIS ORIENTATION | X AXIS ORIENTATION | Y AXIS ORIENTATION



Z AXIS ORIENTATION

- CLEAR PAYLOAD BAY
- WITHIN RMS REACH ENVELOPE
- CLEAR ORBITER RADIATORS
- STRUCTURE TRANSLATION CAPABILITY
- DIRECT VISIBILITY-MARGINAL

X AXIS ORIENTATION

- CLEAR PAYLOAD BAY
- LIMITED RMS REACH
- CLEAR ORBITER RADIATORS
- STRUCTURE TRANSLATION CAPABILITY
- DIRECT VISIBILITY – MARGINAL

Y AXIS ORIENTATION

- PAYLOAD BAY OBSTRUCTION
- LIMITED RMS REACH
- OBSTRUCTS ORBITER RADIATOR
- STRUCTURE TRANSLATION CAPABILITY
- DIRECT VISIBILITY – MARGINAL

Figure A-16. Construction Fixture Orientation Trade

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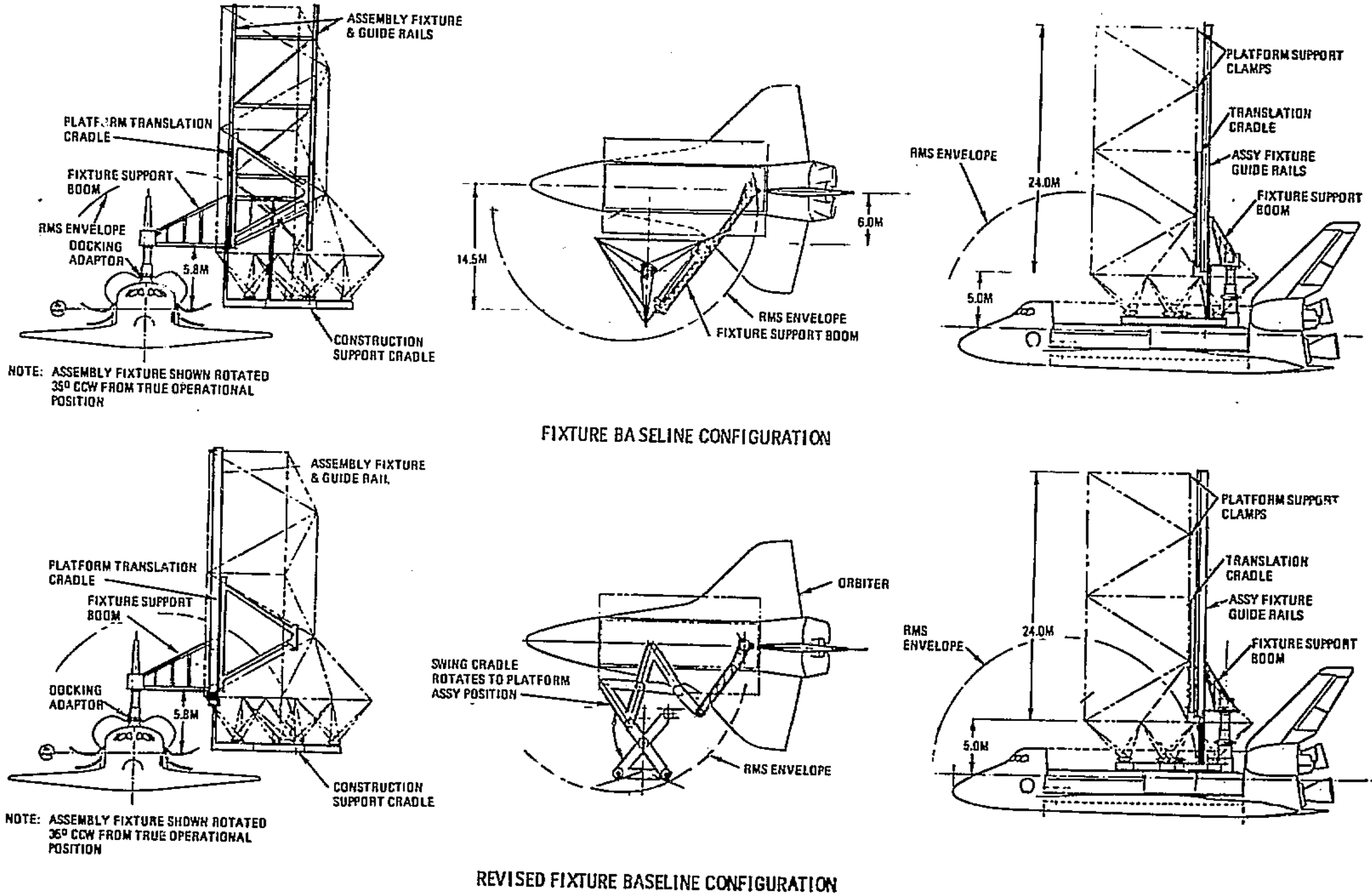

Figure A-17. Erectable Platform Construction Fixture

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